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RESEARCH MEMORANDUM

COMPONENT AND OVER-ALL PERFORMANCE EVALUATION OF AN AXIAL-FLOW TURBOJET ENGINE OVER A RANGE OF ENGINE-INLET REYNOLDS NUMBERS

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SUMMARY

An investigation was conducted in an altitude test chamber at the NACA Lewis laboratory to evaluate the performance of an axial-flow turbojet engine over a range of engine-inlet Reynolds numbers. The range of Reynolds numbers investigated provided data which were applicable over a range of altitudes from 10,000 to 50,000 feet at a flight Mach number of 0.7; 100 percent ram-pressure recovery was assumed.

Reducing the engine inlet Reynolds number resulted in a reduction of corrected air flow and compressor efficiency but did not affect the compressor pressure ratio at a given corrected engine speed. The decreased compressor efficiency required an increase in turbine power that resulted in an increase in exhaust-gas total temperature.

Combustion efficiency is presented as a function of a combustion parameter. At low values of this parameter, which corresponded to low Reynolds number operation for this investigation, combustion efficiency decreased very rapidly. As a result of the combined effects of decreased compressor efficiency and combustion efficiency, the reduction in engine-inlet Reynolds number resulted in an increased fuel flow. At rated corrected engine speed this increase was about 12 percent.

A method is presented whereby conventional performance variables such as net thrust and specific fuel consumption may be obtained for any flight condition within the range of Reynolds numbers investigated. The increased exhaust-gas temperature caused an increase in tail-pipe total pressure which offset the decrease in corrected air flow and thus produced a generalization of the thrust parameter.

INTRODUCTION

In previous altitude-performance evaluations (for example, reference 1), data were obtained over a range of altitudes and flight Mach

[redacted]

numbers and were generalized by the use of the conventional temperature and pressure factors as described in reference 2. Failure of the performance variables to generalize for all altitudes and flight Mach numbers, over the range of engine speeds where sonic flow exists in the exhaust nozzle, has been shown to result from either a Reynolds number effect or a variation in combustion efficiency (reference 3).

The investigation, conducted over a range of altitudes and flight Mach numbers, produced data which were of limited applicability. The present investigation of the component and over-all performance characteristics of an axial-flow turbojet engine of current interest was 2441
therefore conducted over a range of engine-inlet Reynolds numbers at the NACA Lewis laboratory. In order to simplify operation, Reynolds number index, which is proportional to Reynolds number at a given corrected engine speed and is a function only of engine-inlet total pressure and temperature, was used instead of Reynolds number. Because departures from established generalization may be investigated directly and primarily by this method, fewer data are required and are applicable to an extensive range of flight conditions.

Data were obtained over a range of Reynolds number indices from 0.2 to 0.95 which produced data which were applicable over a range of altitudes from 10,000 to 50,000 feet at a flight Mach number of 0.7. Compressor, combustor, turbine, and over-all engine performance data are presented over the range of Reynolds number indices investigated. The trends of over-all engine performance are discussed with reference to the Reynolds number effects on the component performance. Combustion efficiency is presented as a function of combustor-inlet conditions over the range of engine-inlet conditions and engine speeds investigated.

The data obtained in this investigation provide a means of predicting the performance at any flight condition for which critical flow exists in the exhaust nozzle. An example is included to illustrate the method of obtaining conventional performance parameters such as net thrust and specific fuel consumption for a given flight condition from the data presented herein. Because the data presented in this report are valid for only one exhaust-nozzle area, an example of a method of adjusting the performance data for small variations in exhaust-nozzle area is also presented.

APPARATUS

Engine

A J35-A-29 axial-flow turbojet engine which had an 11-stage compressor, a pressure ratio of 5.25:1 at the rated engine speed of 8000 rpm, eight tubular combustion chambers, and a single-stage

turbine (fig. 1), was investigated. This engine was a preproduction model of the J35-A-29 engine and had the same power section as the J35-A-33 engine. A fixed conical exhaust nozzle having a diameter of 18.00 ±0.01 inches at 60° F was installed on the engine. This nozzle is designed to produce a tail-pipe gas temperature of 1300° F (1760° R) at rated engine speed and static sea-level conditions. At these operating conditions and when inlet screens are used, the manufacturer guarantees a rated thrust of 5400 pounds with a specific fuel consumption of 1.06 pounds per hour per pound of thrust and an estimated air flow of
2441 91 pounds per second. The rated thrust without inlet screens would be 5600 pounds. The maximum dimensions of the engine are a 37-inch diameter and a 146-inch length. The dry engine weight without starter generator and tachometer generator is 2305 pounds.

The automatic fuel control for the engine was replaced with an adjustable pressure-control valve to allow a wider range of operation and full throttle sensitivity at high altitude. An aluminum bullet-type accessory cover and bell cowl (fig. 1) were installed at the compressor inlet to obtain a smooth air flow into the compressor.

Altitude Chamber

The altitude chamber in which the engine was installed is 10 feet in diameter and 60 feet long (fig. 2). A honeycomb is installed in the chamber upstream of the test section to straighten and smooth the flow of inlet air. The front bulkhead, which incorporated a labyrinth seal around the forward end of the engine, was used to prevent the flow of combustion air directly into the exhaust system and to provide a means of maintaining a pressure difference across the engine. A 14-inch butterfly valve was installed in the front bulkhead to provide cooling air for the engine compartment. A rear bulkhead was installed to act as a radiation shield and to prevent recirculation of exhaust gases about the engine.

Air is supplied to the inlet section of the engine through a supply line from the laboratory air system. Combustion air can be obtained from this system over a range of temperatures from -70° to 85° F. Small changes in the inlet-air temperature are obtained by the use of electric heaters installed in a bypass line upstream of the chamber. The inlet and exhaust pressures are controlled by means of remote-control valves in the supply lines and the exhaust lines, respectively.

The exhaust gases from the jet nozzle are removed from the exhaust section of the altitude chamber through a diffusing elbow and a dry-type primary cooler. A dry-type secondary cooler downstream of the exhaust valves further cools the hot gases before passing them into the laboratory exhaust system.

INSTRUMENTATION

The locations of the instrumentation stations, before and after each of the principal components of the engine, are shown in figure 3. The detailed arrangement of the separate temperature and pressure probes at each station is shown in figure 4 for those stations at which data are presented herein.

Engine-inlet pressure and temperature were set for a given run according to the readings of the instrumentation at station 1. The temperatures and pressures measured at station 2 were used in calculating the altitude correction factors θ and δ, and the compressor air-flow. (All symbols are defined in the appendix.) Pressure and temperature instrumentation was also installed to determine engine midframe air-bleed. The engine-air flow was equal to the air-flow measured at station 2 minus the air bled off at the midframe. Combustor static pressure was obtained at station 4 from static-pressure taps in combustors 2 and 6. One total-pressure probe was located approximately in the center of each of the eight transitions from the combustor to the turbine at station 5. Pressure and temperature probes at each station, except station 7, were so located that a mean value of temperature or pressure could be obtained directly by averaging the individual readings. At station 7, the average values of total temperature and pressure were obtained from plots of the temperature and pressure profiles, and static pressure was obtained from a mechanical average of four wall-static orifices. These measurements at station 7 were used to calculate jet thrust.

The atmospheric pressure surrounding the jet-nozzle was measured by two probes located near the jet-nozzle outlet in the exhaust portion of the chamber. In order to improve the accuracy of data and the ease of operation, two calibrated aneroid manometers (one high pressure and one low pressure) were used to set inlet and exhaust pressures; and an electrically operated strobotac was installed to assist in maintaining constant engine-speed settings.

Fuel flow was measured by two rotameters connected in series; two rotameters were necessary to cover the entire range of flows and to keep the physical size at a minimum. The rotameters were calibrated with the fuel used in the investigation (MIL-F-5624A, grade JP-3).

PROCEDURE

Reynolds number index, a function only of temperature and pressure, is defined by the expression $\delta_2/\phi_2\sqrt{\theta_2}$. The derivation of this expression is presented in reference 4 where δ_2 is the ratio of compressor-inlet absolute total pressure to absolute total pressure of NACA standard atmosphere at sea level; θ_2 is the ratio of compressor-inlet absolute total temperature to absolute total temperature of NACA standard atmosphere at sea level; and ϕ_2 is the ratio of viscosity at the compressor-inlet total temperature to viscosity at NACA standard sea-level temperature.

The inlet conditions were varied to correspond to Reynolds number indices from 0.2 to 0.95. For each inlet condition the exhaust pressure was reduced to the minimum of the exhaust system with the engine operating at rated speed. The inlet temperature and pressure and the exhaust pressure were then maintained while data were taken over a range of engine speeds from rated speed to approximately the speed where the exhaust nozzle was barely choked. A summary of the operating conditions covered in the investigation is given in the following table:

Reynolds number index	Inlet total temperature (°R)	Inlet total pressure (lb/sq ft)	Ram-pressure ratio
0.2	405	310	1.45
.3	405	465	1.4
.4	405	610	1.4
.4	465	740	1.78
.4	465	740	2.18
.5	465	923	1.78
.5	465	923	2.22
.6	465	1100	1.78
.95	520	2000	2.32

As shown in the table, three ram-pressure ratios P_2/p_0 were used at a Reynolds number index of 0.4 and two at 0.5 to verify the generalization with exhaust-pressure variation. At a Reynolds number index of 0.4, two sets of inlet conditions were used to determine whether there were any effects of temperature and pressure other than those of Reynolds number variations.

RESULTS AND DISCUSSION

The performance data obtained in this investigation were corrected to standard sea-level conditions in the conventional manner (reference 2) and are presented in table I. Generalization of data for various engine-inlet conditions that give the same Reynolds number index requires choked flow in the exhaust nozzle. The range of engine speeds over which the exhaust nozzle of the engine was choked is shown in figure 5 for a range of altitudes and flight Mach numbers. At all altitudes, this minimum corrected engine speed at which choking occurred decreased linearly from about 7750 rpm at a flight Mach number of 0.2 to about 5450 rpm at a flight Mach number of 1.3. The data of this report may be used to determine performance only at flight conditions in the choked region above this curve.

In order to aid in determining the Reynolds number index corresponding to a given flight condition and thereby determine the engine performance at NACA standard altitude conditions from the generalized data presented, the values of δ, θ, ϕ, and Reynolds number index are given in table II for a wide range of flight conditions; 100 percent ram-pressure recovery was assumed. 2441

Compressor Performance

Compressor performance characteristics are presented in figure 6 for the range of Reynolds number indices investigated. The decrease in compressor efficiency encountered with the reduction in Reynolds number index is shown in figure 6(a). The peak compressor efficiency occurred at a corrected engine speed of about 7000 rpm for all Reynolds number indices investigated and decreased from 82 percent to 78 percent as Reynolds number index was decreased from 0.95 to 0.2 (corresponding to an increase in altitude from 10,000 to 50,000 ft at a flight Mach number of 0.7). Corrected compressor air flow is shown as a function of corrected engine speed over the range of Reynolds number indices investigated in figure 6(b). At Reynolds number indices of 0.4 and above, corrected air flow generalized at corrected engine speeds below about 7200 rpm. At a corrected engine speed of 8000 rpm, the corrected compressor air flow decreased from 90.7 to 86.0 pounds per second as Reynolds number index was decreased from 0.95 to 0.2. The ratio of midframe air-bleed to compressor air flow is presented in figure 7 as a function of the ratio of compressor outlet total pressure to ambient static pressure. The engine air flow is equal to the compressor air flow minus the midframe air-bleed. The decrease in efficiency and air flow will shift the compressor operating point (equilibrium point with the turbine) because of the increase in work required of the turbine. The amount of this shift is illustrated in figure 6(c). Although the compressor operating lines shifted as Reynolds number index was reduced, the simultaneous decrease in air flow and increase in turbine-inlet temperature due to the loss in efficiency was such that the variation of compressor pressure ratio with corrected engine speed generalized for all Reynolds number indices investigated as shown in figure 6(d).

Combustor Performance

Variation of the total-pressure-loss ratio across the combustor with corrected engine speed is shown in figure 8(a). Over the range of engine speeds investigated, the total-pressure-loss ratio decreased with increasing corrected engine speed with no apparent effects of Reynolds number index. Combustion efficiency (fig. 8(b)) was found to generalize with the parameter $p_4{}^2/W_{a,3}$ which is proportional to a combustion parameter derived in reference 5. At a combustion parameter value above 300,000, the correlation was within ± 0.025 and a constant combustion efficiency of about 0.94 was indicated. At lower values of the combustion parameter, combustion efficiency dropped rapidly and the data scatter was approximately doubled. From these data, it is concluded that for the Reynolds number indices and the corrected engine speeds of this investigation, the effects of fuel-air ratio and fuel-spray pattern were secondary. The dashed curve in figure 8(b) shows the results of a subsequent investigation (reference 6) in which modified combustor liners had been installed and indicates an improvement in combustion efficiency of about 0.03 over a large part of the operable range. These modified liners are designated "smokeless" liners by the manufacturer and are standard on the production models of the engine.

In order to use figure 8(b) for predicting combustion efficiency under altitude operating conditions, the variation of the term $p_4{}^2/W_{a,3}$ with corrected engine speed and Reynolds number index must be evaluated. In order to facilitate this evaluation, a generalized plot of $p_4{}^2/W_{a,3}\delta\sqrt{\theta}$ against corrected engine speed for the various Reynolds number indices is presented in figure 8(c).

Turbine Performance

Turbine total-pressure ratio (fig. 9(a)) generalized for all conditions at the high corrected engine speeds but the data scattered at the lower engine speeds investigated. Turbine efficiency (fig. 9(b)) remained nearly constant at about 0.81 over the engine-speed range investigated at Reynolds number indices above 0.4. Reducing the Reynolds number index to 0.3 and 0.2 lowered the efficiency by 0.01 and 0.03, respectively. The corrected turbine gas flow (fig. 9(c)) was constant over the range of corrected engine speeds investigated and no Reynolds number effect was obtained. The constant corrected gas flow resulted from the turbine nozzle being choked over this range of engine speeds and indicates that within the accuracy of the data there was no apparent reduction in effective turbine-nozzle area due to a Reynolds number effect. The decrease in turbine efficiency at the low Reynolds number indices is believed to have been the result of a shift in the turbine operating point and a Reynolds number effect.

Generalized Engine Performance

The effect of Reynolds number index on generalized engine performance is shown in figure 10. Corrected exhaust-gas total temperature (fig. 10(a)) increased with a reduction in Reynolds number index. The increase at any given corrected engine speed was 130° R with a reduction in Reynolds number index from 0.5 to 0.2. This increase in corrected exhaust-gas total temperature is the result of a shift in the engine operating point caused primarily by the decrease in compressor efficiency and turbine efficiency. At corrected engine speeds above about 7000 rpm, the corrected fuel flow (fig. 10(b)) generalized for Reynolds number indices from 0.95 to 0.4 and increased at the lower values of Reynolds number index. This increase resulted from the required rise in exhaust-gas temperature and the decrease in combustion efficiency at low Reynolds number indices. The increase in corrected fuel flow at rated corrected engine speed was approximately 12 percent as Reynolds number index was reduced from 0.4 to 0.2.

The effect of Reynolds number index on the engine pumping characteristics is illustrated in figure 10(c), which also includes lines of constant corrected engine speed. These constant-speed lines were obtained from figure 10(a) by using the relation that the engine total-temperature ratio is equal to the corrected exhaust-gas total temperature divided by NACA sea-level standard temperature. As the Reynolds number index was reduced, the pumping-characteristic curves shifted in the direction of increased engine total-temperature ratio at a given engine total-pressure ratio. This shift in the curves reflects the reduced efficiency of the compressor and turbine. At a given corrected engine speed, a decrease in the Reynolds number index resulted in an increase in engine total-temperature ratio with an accompanying slight rise in engine total-pressure ratio. The combined effect of these increases in temperature and pressure ratios and the decreased air flow was such as to produce no apparent Reynolds number effect on the corrected jet-thrust parameter (fig. 10(d)). The jet-thrust parameter therefore generalized throughout the range of the Reynolds number indices and corrected engine speeds investigated.

The conventional performance variables such as net thrust and specific fuel consumption can be obtained for any flight condition directly from the data which have been presented. An example is presented in the appendix to indicate the technique for thus applying the data.

The following comparison has been made between the manufacturer's rated values for sea-level static conditions, the manufacturer's calibration, and the results of this investigation by utilizing this technique of transforming the data:

	Manufacturer's rated values	Manufacturer's calibration with afterburner	Present investigation	Investigation corrected to exhaust-gas total temperature of 1300° F
Engine speed (rpm)	8000	8000	8000	8000
Thrust (lb)	5400	5250	5100	5465
Specific fuel consumption, $\left(\frac{\text{lb/hr}}{\text{lb/thrust}}\right)$	1.06	1.09	1.11	1.05
Turbine-outlet total temperature (°F)	[a]1340	1285	1245	1300
Air flow (lb/sec)	[b]89	----	89.8	89.8

[a]Maximum permissable operating temperature; normal limiting temperature is 1300° F for an inlet temperature of 60° F.

[b]Estimated.

The performance values in the second column were obtained by the manufacturer during a sea-level calibration of the engine used for the present investigation. This performance calibration by manufacturer was made with a short afterburner installed on the engine and at a turbine-outlet total temperature of only 1285° F as compared with 1300° F specified for the performance. Consequently, the thrust obtained was about 3 percent below the rated value. Had the turbine-outlet temperature been between 1300° and 1340° F, the thrust would have exceeded the manufacturer's rated value.

The values shown in the third column of the table were taken directly from the data presented herein. Because the turbine-outlet temperature was only 1245° F, the thrust fell nearly 6 percent below the rated value and the specific fuel consumption was nearly 5 percent above the rated value. These performance values were adjusted to a turbine-outlet temperature of 1300° F by means of the engine pumping characteristics and the results are shown in the last column of the table. The technique utilized to adjust the performance for small changes in exhaust-gas temperature and consequently exhaust-nozzle area is explained in the appendix. This adjusted performance met the manufacturer's rated values.

CONCLUDING REMARKS

An investigation of an axial-flow turbojet engine has shown that reducing the engine-inlet Reynolds number has a detrimental effect on engine performance. The Reynolds number variation investigated corresponds to a variation in altitude from 10,000 to 50,000 feet at a flight Mach number of 0.7. A reduction in engine-inlet Reynolds number produced a reduction in compressor efficiency and air flow but did not affect compressor pressure ratio at a given corrected engine speed. The reduced compressor efficiency required an increase in turbine power for each pound of air handled. This power increase was accompanied by an increase in engine fuel-air ratio and an attendant increase in turbine-inlet temperature. At low engine-inlet Reynolds numbers a slight decrease in turbine efficiency occurred, which is attributed to the combined effect of the shift in turbine operating point and the reduced Reynolds number. The higher turbine temperature obtained at low engine-inlet Reynolds numbers produced a higher corrected tail-pipe pressure; together, these offset the decrease in corrected air flow and thereby resulted in a generalization of the thrust parameter.

Combustion efficiency was presented as a function of a combustion parameter which is based on the assumption that efficiency is proportional to combustor-inlet pressure and temperature and inversely proportional to inlet velocity. At low values of this parameter, which in general correspond to low engine-inlet Reynolds number operation, the combustion efficiency dropped very rapidly. The combined effects of decreased combustion efficiency and compressor efficiency, resulted in an increase in engine fuel consumption at low inlet Reynolds numbers. At rated corrected speed this increase amounted to about 12 percent.

The presentation of combustion efficiency as a function of combustor-inlet conditions and of engine performance parameters as a function of Reynolds number allows the determination of performance of this engine for any flight condition within the limits of the engine-inlet Reynolds numbers investigated.

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APPENDIX - CALCULATIONS

The following symbols are used in this report:

A	area, sq ft
$C_{V,e}$	effective velocity coefficient, $\frac{\text{actual } V_e}{\text{ideal } V_e}$
c_p	specific heat at constant pressure
F	net thrust, lb
F_j	jet thrust, lb
f	fuel-air ratio
g	acceleration due to gravity, ft/sec^2
H	enthalpy, Btu/lb
M	flight Mach number
N	engine speed, rpm
P	total pressure, lb/sq ft
p	static pressure, lb/sq ft
R	gas constant, ft-lb/(lb)($^{\circ}$R)
T	total temperature, $^{\circ}$R
t	static temperature, $^{\circ}$R
V	velocity, ft/sec
V_e	effective velocity, $V_j + \frac{A_j g}{(1+f)W_{a,3}} (p_j - p_0)$, ft/sec
W_a	air flow, lb/sec
W_f	fuel flow, lb/hr
W_g	gas flow, lb/sec
γ	ratio of specific heats
δ	ratio of total pressure to NACA standard sea-level pressure, 2116 lb/sq ft

η_b combustion efficiency

η_c compressor efficiency

η_t turbine efficiency

θ ratio of total temperature to NACA standard sea-level temperature, 519° R

ϕ ratio of coefficient of viscosity corresponding with total temperature to coefficient of viscosity corresponding with NACA standard sea-level temperature, 519° R. This ratio is a function only of temperature and is equal to $\dfrac{735\theta^{1.5}}{T+216}$ 2447

Subscripts:

m midframe

n nozzle

j jet

0 ambient

1 bellmouth inlet

2 compressor inlet

3 compressor outlet or combustor inlet

4 combustor

5 combustor outlet or turbine inlet

6 turbine outlet

7 exhaust-nozzle inlet

Methods of Calculation

Jet-thrust parameter. - The jet thrust was determined from the measured air flow, the measured total-to static pressure ratio at the exhaust nozzle, and an estimated effective velocity coefficient of 0.98, by using an effective velocity parameter which, for critical flow, is equivalent to

$$F_j = C_{V,e}\left\{(1+f)\ W_{a,3}\sqrt{\frac{2R}{g}\ \frac{\gamma_7}{\gamma_7-1}\ T_7\left[1 - \left(\frac{2}{\gamma+1}\right)\right]} + A_j(p_j-p_0)\right\}$$

where A_j is the effective throat area of the jet and p_j is the static pressure at that location. That is,

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$$p_j = P_7\left(\frac{2}{\gamma+1}\right)^{\frac{\gamma}{\gamma-1}}$$

and

$$A_j = \left(\frac{\sqrt{T_7}}{P_7}\right)\frac{(1+f)W_{a,3}}{\sqrt{\frac{\gamma g}{R}}}\left(1 + \frac{\gamma_7-1}{2}\right)^{\frac{\gamma+1}{2(\gamma-1)}}$$

The jet-thrust parameter is defined by the expression $(F_j+p_0A_n)/\delta_2$, where A_n is the hot area of the exhaust nozzle obtained from the cold area, measured skin temperature, and the coefficient of expansion of the material. When using the jet-thrust-parameter curve to find jet thrust, it is suggested that an average hot area of 1.78 square feet be used instead of the cold area of 1.765 square feet for A_n.

Correcting test values to flight conditions. - Let it be assumed that it is desired to determine the performance of this engine at an NACA standard altitude of 40,000 feet, a flight Mach number of 0.6, and an actual engine speed of 7500 rpm. Values of δ_2, θ_2, and Reynolds number index of 0.2364, 0.8118, and 0.3065, respectively, can be obtained from table II.

Then the corrected engine speed is

$$\frac{N}{\sqrt{\theta_2}} = \frac{7500}{\sqrt{0.8118}} = \frac{7500}{0.9010} = 8324 \text{ rpm}$$

The values for the various parameters will be taken from the curves in this report at this corrected engine speed and a Reynolds number index of 0.3. From figure 6(c), corrected compressor air flow $W_{a,2}\sqrt{\theta_2}/\delta_2$ equals 89.4 pounds per second. Then actual compressor air flow is

$$W_{a,2} = \left(\frac{W_{a,2}\sqrt{\theta_2}}{\delta_2}\right)\frac{\delta_2}{\sqrt{\theta_2}} = \frac{89.4(0.2364)}{0.9010} = 23.5 \text{ lb/sec}$$

[redacted]

At a Mach number of 0.6, the ram-pressure ratio P_2/p_0 is 1.276. In figure 6(d), the compressor pressure ratio P_3/P_2, is given as 5.5; this gives

$$\frac{P_3}{p_0} = \left(\frac{P_2}{p_0}\right)\left(\frac{P_3}{P_2}\right) = 7.01$$

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In figure 7, the ratio of the midframe air bleed to compressor air flow is 0.0185. Therefore, the engine air flow is

$$W_{a,3} = 0.9815\ W_{a,2} = 23.1\ \text{lb/sec}$$

In figure 8(c), the corrected combustion parameter $p_4{}^2/W_{a,3}\ \delta_2\sqrt{\theta_2}$ is 146×10^4. The combustion parameter is

$$\frac{p_4{}^2}{W_a} = \frac{p_4{}^2}{W_{a,3}\delta_2\sqrt{\theta_2}}\left(\delta_2\sqrt{\theta_2}\right) = 146\times10^4\times0.2364\times0.901 = 311{,}000$$

In figure 8(b), this value corresponds to a combustion efficiency of 0.94.

In figure 10(a), corrected exhaust-gas total temperature T_7/θ_2 equals 1915° R and the actual exhaust-gas total temperature is

$$T_7 = 1915\times0.8118 = 1555^\circ\ \text{R}$$

In figure 10(b), corrected fuel flow $W_f/\delta\sqrt{\theta}$ is given as 6600 pounds per hour. Then the actual fuel flow is

$$W_f = (6600)(0.2364)(0.9010) = 1406\ \text{lb/hr}$$

In figure 10(d), the jet-thrust parameter $(F_j+p_0A_n)/\delta_2$ is 9350 pounds. The jet thrust is

$$F_j = 9350\ \delta_2 - p_0A_n = 9350\times0.2364-391.9\times1.78 = 1512\ \text{lb}$$

where p_0 is 391.9 pounds per square foot, the static pressure at 40,000 feet, and A_n is 1.78 square feet as explained previously. At an altitude of 40,000 feet the speed of sound is 971 feet per second and a Mach number of 0.6 corresponds to an air speed of 583 feet per second. The inlet momentum is, therefore,

$$\frac{W_{a,2}}{g}V_0 = \frac{23.5}{32.17}(583) = 425\ \text{lb}$$

and the net thrust is

$$F = F_j - \frac{W_{a,2}}{g} V_0 = 1512\text{-}425 = 1087 \text{ lb}$$

Net thrust specific fuel consumption is

$$\frac{W_f}{F} = \frac{1406}{1087} = 1.29 \text{ lb/(hr)(lb thrust)}$$

2441 Test values corrected to exhaust-gas total temperature of 1300° F. - In RESULTS AND DISCUSSION values are given for rated engine conditions, which were corrected to an exhaust-gas total temperature of 1300° F (1760° R) instead of the 1245° F (1705° R) obtained from figure 10(a) for standard sea-level conditions. This difference in temperature was due to a ram-pressure-ratio effect which becomes negligible above a ram pressure ratio P_1/p_0 of 1.4. This ram-pressure-ratio effect is due to a variation in the effective exhaust-nozzle area or flow coefficient with nozzle pressure ratio. The method used to adjust the rated performance at standard sea-level conditions, which is given in the following discussion, applies equally well for similar small changes in turbine-outlet temperature or effective exhaust-nozzle area at any other flight conditions.

At rated engine speed and standard sea-level static conditions, the maximum exhaust-gas total temperature should have been 1760° R; the resulting engine total-temperature ratio T_7/T_2 would have been 1760/519 or 3.39. If the pumping-characteristic curve (fig. 10(c)) is entered at a Reynolds number index of 0.95 and at this value of temperature ratio, the engine total-pressure ratio P_7/P_2 is 2.01. Exhaust-nozzle-inlet total pressure is then found to be

$$P_7 = \frac{P_7}{P_2} P_2 = (2.01)(2116) = 4253 \text{ lb/sq ft}$$

It is assumed that the small change in engine total-pressure ratio from the actual value of 1.96 to the new value of 2.01 does not change the compressor operating point enough to appreciably change the air flow. The fuel flow, air flow, and engine total-temperature ratios can be found for the actual operating conditions at the desired inlet conditions (in this case standard sea-level temperature and pressure) over a range of engine speeds by using the technique explained in the previous example. With these values, a plot can be made of fuel-air ratio against engine total-temperature ratio, as shown in figure 11. This plot yields the fuel-air ratio for the desired engine temperature ratio, in this case, $f = 0.0177$. This curve was based on the assumptions that T_2 was constant and that for a small change in T_7/T_2, c_p and η_b were constant. The ratio of specific heats γ_7 of the exhaust gases was taken consistent with temperature and fuel-air ratio (reference 7). In this case, with $T_7 = 1760°$ R and $f = 0.0177$, γ equals 1.323.

Midframe air bleed was obtained from figure 7, in this case $W_{a,2} = 90.7$ pounds per second; $W_{a,3} = W_{a,2} - 0.01\ W_{a,2} = 89.8$ pounds per second. If these values are used in the equations previously discussed,

$$A_j = \left(\frac{\sqrt{1760}}{4253}\right)\frac{(1.0177)(89.8)}{\sqrt{\frac{(1.323)(32.17)}{53.4}}}\left(1 + \frac{1.323-1}{2}\right)^{\frac{1.323+1}{2(1.323-1)}} = 1.695 \text{ sq ft}$$

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$$p_j = 4253\left(\frac{2}{1.323+1}\right)^{\frac{1.323}{0.323}} = 2297 \text{ lb/sq ft}$$

and

$$F_j = 0.98\left\{(1.0177)(89.8)\sqrt{\frac{(2)(53.4)}{32.17}\,\frac{1.323}{0.323}\,(1760)\left[1 - \left(\frac{2}{2.323}\right)\right]} + 1.695(2297-2116)\right\} = 5465 \text{ lb}$$

Fuel flow was determined from the relation

$$W_f = (f)(W_a)(3600) = 0.0177(89.8)(3600) = 5715 \text{ lb/hr}$$

and net thrust specific fuel consumption was determined as

$$\frac{W_f}{F} = \frac{5715}{5465} = 1.045 \text{ lb/(hr)(lb thrust)}$$

In correcting data of this report at any flight condition for the effect of such a change in nozzle size:

(1) Assume that corrected air flow plotted against corrected engine speed does not change with small changes in exhaust-gas pressure and temperature.

(2) Plot fuel-air ratio against engine temperature ratio as explained.

(3) The family of curves of corrected exhaust-gas total temperature against corrected engine speed should be so adjusted that, at a speed of 8000 rpm and a Reynolds number index of 0.95, the corrected exhaust-gas total temperature is 1760° R. The new temperature-speed curves should be drawn parallel to and spaced the same as the curves in figure 10(a).

(4) Thrust, fuel flow, and specific fuel consumption can be determined throughout the range of engine speeds by using this new curve together with the engine pumping characteristics, as shown by the example for rated speed at sea-level static conditions.

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REFERENCES

1. Vincent, K. R., and Gale, B. M.: Altitude Performance of J35-A-17 Turbojet Engine in an Altitude Chamber. NACA RM E50I15, 1950.

2. Sanders, Newell, D.: Performance Parameters for Jet-Propulsion Engines. NACA TN 1106, 1946.

3. Fleming, William A.: Effects of Altitude on Turbojet Engine Performance. NACA RM E51J15, 1951.

4. Sanders, Newell D., and Behun, Michael: Generalization of Turbojet-Engine Performance in Terms of Pumping Characteristics. NACA TN 1927, 1949.

5. Childs, J. Howard: Preliminary Correlation of Efficiency of Aircraft Gas-Turbine Combustors for Different Operating Conditions. NACA RM E50F15, 1950.

6. Braithwaite, Willis M., and Renas, Paul E.: Comparison of Turbojet-Engine Altitude Performance Characteristics and Ignition Limits with MIL-F-5624A Fuel, Grades JP-3 and JP-4. NACA RM E51L05, 1952.

7. Pinkel, Benjamin, and Turner, L. Richard: Thermodynamic Data for the Computation of the Performance of Exhaust-Gas Turbines. NACA ARR 4B25, 1944.

TABLE I - PERFORMANCE AT VARIOUS

Run	Reynolds number index $\frac{\delta}{\phi\sqrt{\theta}}$	Ram pressure ratio $\frac{P_2}{p_0}$	Compressor-inlet total pressure P_2 $\left(\frac{lb}{sq\ ft}\right)$	Compressor inlet total temperature T_2 (^{o}R)	Compressor total pressure ratio $\frac{P_3}{P_2}$	Compressor discharge total temperature T_3 (^{o}R)	Combustor total pressure loss ratio $\frac{P_3-P_5}{P_3}$	Corrected combustor static pressure $\frac{p_4}{\delta_2}$ $\left(\frac{lb}{sq\ ft}\right)$	Calculated turbine inlet total temperature T_5 (^{o}R)	Turbine total pressure ratio $\frac{P_5}{P_6}$	Engine total temperature ratio $\frac{T_7}{T_2}$
1	0.204	1.462	313.9	408	2.646	587	0.0497	5,350	1060	2.337	2.12
2	.199	1.526	311.1	412	2.969	606	.0503	5,992	1090	2.450	2.15
3	.201	1.513	313.8	412	3.349	627	.0438	6,777	1180	2.506	2.32
4	.206	1.438	315.4	406	3.424	626	.0454	6,922	1240	2.435	2.52
5	.205	1.437	314.3	406	4.397	683	.0412	8,956	1520	2.446	3.13
6	.206	1.434	315.7	406	5.112	725	.0366	10,460	1740	2.453	3.59
7	.204	1.415	312.8	407	5.636	761	.0363	11,510	1900	2.486	3.92
8	.202	1.422	310.4	408	6.066	807	.0356	12,370	2050	2.527	4.26
9	0.316	1.380	473.6	400	2.815	580	0.0435	5,706	1040	2.355	2.11
10	.306	1.384	463.6	403	3.497	622	.0457	7,079	1205	2.481	2.39
11	.305	1.405	462.6	403	4.589	682	.0396	9,328	1505	2.494	3.06
12	.304	1.403	463.8	405	5.136	718	.0348	10,490	1665	2.498	3.41
13	.306	1.402	463.7	403	5.605	751	.0346	11,440	1880	2.504	3.84
14	.305	1.420	461.9	403	6.021	794	.0345	12,290	2035	2.540	4.18
15	0.406	2.190	737.9	462	1.859	601	0.0561	3,702	775	2.394	1.35
16	.404	1.782	740.9	465	2.327	637	.0481	4,684	957	2.421	1.66
17	.406	2.187	738.2	462	2.293	632	.0526	4,580	918	2.486	1.61
18	.408	1.413	628.8	408	2.642	580	.0439	5,350	995	2.328	1.99
19	.401	1.776	739.3	467	2.895	678	.0430	5,861	1143	2.507	1.95
20	.409	1.414	625.9	406	3.395	620	.0456	6,876	1160	2.452	2.35
21	.404	2.155	736.3	463	3.391	706	.0465	6,856	1315	2.499	2.27
22	.401	1.777	740.0	467	3.739	735	.0434	7,575	1440	2.516	2.47
23	.412	1.418	623.7	403	4.483	675	.0404	9,101	1465	2.508	2.93
24	.406	1.776	739.5	463	4.824	795	.0367	9,823	1750	2.512	3.11
25	.404	2.161	732.0	461	4.967	799	.0369	10,130	1785	2.507	3.19
26	.412	1.421	621.6	402	5.360	729	.0348	10,940	1725	2.526	3.49
27	.405	1.784	742.3	465	5.513	855	.0347	11,250	2055	2.522	3.61
28	.406	2.164	732.7	460	5.654	855	.0350	11,530	2095	2.524	3.72
29	.410	1.405	621.6	403	5.952	790	.0351	12,120	2015	2.541	4.10
30	0.508	2.241	924.0	462	2.010	609	0.0533	4,007	799	2.475	1.38
31	.504	2.231	922.8	465	2.289	632	.0516	4,572	880	2.511	1.53
32	.508	1.780	926.1	463	2.600	653	.0449	5,243	1040	2.486	1.79
33	.510	1.783	928.9	463	2.809	666	.0433	5,679	1110	2.494	1.89
34	.504	2.202	920.3	464	3.429	708	.0450	7,098	1320	2.512	2.26
35	.511	1.779	918.2	458	3.870	728	.0414	7,847	1470	2.523	2.54
36	.508	1.783	919.9	461	4.733	787	.0374	9,630	1735	2.512	3.06
37	.512	2.202	916.9	457	4.930	788	.0367	10,040	1765	2.509	3.16
38	.504	2.212	913.1	461	5.636	851	.0350	11,490	2095	2.517	3.67
39	.507	1.796	924.0	463	5.602	853	.0342	11,430	2105	2.523	3.67
40	0.604	1.774	1116	468	2.202	631	0.0501	4,412	885	2.369	1.56
41	.599	1.764	1107	468	2.829	673	.0428	5,719	1115	2.490	1.88
42	.602	1.774	1106	466	4.120	752	.0413	8,355	1580	2.508	2.67
43	.598	1.763	1100	466	5.193	821	.0359	10,580	1910	2.507	3.31
44	.598	1.777	1100	467	5.553	854	.0350	11,320	2095	2.516	3.60
45	.598	1.860	1100	466	5.581	857	.0347	11,380	2115	2.510	3.63
46	.599	1.543	1094	464	5.582	855	.0360	11,370	2095	2.511	3.61
47	0.962	2.341	2020	516	2.015	679	0.0523	4,015	869	2.496	1.34
48	.944	2.344	2018	523	2.137	697	.0522	4,264	923	2.516	1.41
49	.959	2.334	2014	516	2.974	749	.0419	6,018	1280	2.521	1.93
50	.938	2.323	2000	522	4.209	844	.0408	8,537	1790	2.502	2.70
51	.948	2.377	2000	518	5.083	899	.0362	10,360	2060	2.505	3.19
52	.953	2.328	2006	517	5.247	910	.0363	10,690	2130	2.502	3.29

NACA

ENGINE OPERATING CONDITIONS

Engine total pressure ratio $\frac{P_7}{P_2}$	Corrected engine speed $\frac{N}{\sqrt{\theta_2}}$ (rpm)	Corrected compressor air flow $\frac{W_{a,2}\sqrt{\theta_2}}{\delta_2}$ $\left(\frac{lb}{sec}\right)$	Compressor efficiency η_c	Combustor efficiency η_b	Corrected combustion parameter $\frac{P_4^2}{W_{a,3}\delta\sqrt{\theta}}$	Corrected turbine gas flow $\frac{W_{g,5}\sqrt{\theta_5}}{\delta_5\sqrt{\gamma_5/1.4}}$ $\left(\frac{lb}{sec}\right)$	Turbine efficiency η_t	Corrected fuel flow $\frac{W_{f,e}}{\delta_2\sqrt{\theta_2}}$ $\left(\frac{lb}{hr}\right)$	Corrected exhaust-gas total temperature $\frac{T_7}{\theta_2}$ (°R)	Corrected jet thrust parameter $\frac{F_j+p_0A_j}{\delta_2}$ (lb)	Run
1.054	6147	57.7	0.731	0.753	49.62×10^4	37.2	0.790	2100	1100	4,760	1
1.121	6366	62.6	.778	.733	57.40	36.5	.792	2390	1115	5,110	2
1.243	6615	67.5	.790	.810	68.01	36.2	.789	2710	1205	5,690	3
1.304	6674	68.1	.777	.940	70.34	37.1	.786	2700	1305	5,980	4
1.668	7354	78.3	.772	.942	103.6	36.8	.790	4390	1625	7,600	5
1.942	7919	85.3	.754	.921	130.5	36.8	.788	6030	1865	8,830	6
2.119	8442	88.3	.733	.898	152.97	36.1	.789	7280	2030	9,560	7
2.251	9025	88.8	.687	.881	176.5	35.2	.799	8420	2205	10,060	8
1.118	6221	61.0	0.766	0.868	53.37	36.7	0.809	1890	1095	5,030	9
1.305	6723	69.9	.794	.927	71.89	36.4	.807	2540	1240	5,970	10
1.713	7459	83.0	.788	.946	106.2	36.9	.800	4440	1585	7,890	11
1.926	7935	86.9	.772	.929	128.6	36.3	.799	5600	1770	8,760	12
2.104	8488	90.0	.737	.921	148.17	36.6	.791	6990	1990	9,610	13
2.234	9089	90.8	.689	.907	170.0	35.9	.803	8100	2165	10,130	14
.7214	5508	49.8	0.647	0.655	27.53	36.7	0.824	651	702	3,260	15
.8977	5881	56.9	.738	.841	38.9	36.6	.816	1060	861	4,070	16
.8563	5891	57.0	.729	.859	36.8	37.1	.812	957	834	3,950	17
1.067	6074	59.8	.761	.898	47.90	37.1	.814	1580	1030	4,800	18
1.076	6329	65.4	.786	.948	53.19	36.6	.814	1580	1010	4,950	19
1.289	6650	70.3	.794	.943	67.81	37.1	.805	2450	1220	5,910	20
1.257	6668	70.9	.793	.968	67.01	36.9	.814	2280	1175	5,770	21
1.382	6911	75.3	.794	.947	77.52	36.8	.813	2830	1280	6,370	22
1.667	7416	82.7	.793	.957	101.5	36.8	.806	4100	1520	7,680	23
1.799	7669	86.8	.788	.949	113.5	36.8	.809	4790	1610	8,250	24
1.855	7729	86.8	.788	.941	120.7	36.3	.807	5050	1655	8,380	25
1.992	8227	90.5	.757	.947	134.5	36.6	.811	5930	1810	9,160	26
2.058	8459	91.6	.744	.941	141.3	36.7	.811	6470	1875	9,420	27
2.105	8512	91.1	.742	.932	149.6	36.1	.806	6800	1930	9,510	28
2.210	9085	92.1	.691	.933	162.9	36.6	.808	7780	2125	10,140	29
.7545	5638	52.4	0.693	0.744	30.65	36.0	0.822	634	714	3,420	30
.8451	5867	57.6	.742	.875	36.31	36.7	.821	841	792	3,890	31
.9800	6106	61.6	.763	.924	45.21	36.7	.812	1160	927	4,490	32
1.050	6276	64.6	.783	.941	50.55	36.7	.816	1470	982	4,830	33
1.268	6695	71.7	.803	.971	71.20	36.7	.815	2250	1170	5,810	34
1.432	6995	76.8	.799	.949	81.67	36.7	.809	3040	1320	6,590	35
1.762	7604	86.1	.787	.956	110.1	36.9	.811	4620	1585	8,100	36
1.839	7731	87.2	.793	.952	118.0	36.5	.806	4940	1635	8,360	37
2.101	8462	91.5	.752	.929	147.9	36.1	.808	6680	1905	9,480	38
2.088	8478	92.2	.752	.931	145.5	36.6	.801	6730	1905	9,540	39
.8690	5759	54.9	0.729	0.865	35.44	36.6	0.833	849	810	3,900	40
1.059	6272	64.1	.789	.958	51.67	36.0	.824	1420	977	4,810	41
1.530	7133	78.7	.810	.944	90.53	36.2	.809	3390	1385	6,950	42
1.937	7952	89.7	.784	.951	127.8	36.4	.808	5480	1715	8,820	43
2.069	8424	91.3	.755	.950	143.8	36.2	.809	6340	1870	9,390	44
2.087	8467	91.3	.752	.945	145.5	36.1	.812	6440	1880	9,430	45
2.086	8488	91.6	.747	.934	144.6	36.3	.817	6510	1875	9,470	46
.7510	5646	53.9	0.699	0.938	30.29	36.0	0.830	456	695	3,410	47
.7879	5744	56.1	.729	.942	32.89	36.3	.827	563	729	3,600	48
1.098	6395	67.0	.806	.965	55.04	36.1	.818	1560	1000	5,010	49
1.568	7200	80.5	.818	.949	92.74	36.4	.809	3570	1400	7,110	50
1.900	7856	89.3	.796	.942	123.3	36.4	.813	5280	1655	8,600	51
1.960	8030	90.7	.789	.938	129.3	36.4	.816	5670	1705	8,880	52

TABLE II - REYNOLDS NUMBER INDEX VARIATION WITH

FLIGHT MACH NUMBER AND ALTITUDE

[100 Percent ram-pressure recovery assumed.]

Altitude (ft)	Flight Mach number M_0	Pressure ratio δ	Temperature ratio θ	Viscosity ratio ϕ	Reynolds number index $\delta/\phi\sqrt{\theta}$
0	0	1.000	1.000	1.000	1.000
	.1	1.007	1.002	1.002	1.004
	.2	1.028	1.008	1.006	1.018
	.3	1.064	1.018	1.013	1.041
	.4	1.117	1.032	1.023	1.075
	.5	1.186	1.050	1.036	1.117
	.6	1.276	1.072	1.051	1.173
	.7	1.387	1.098	1.069	1.238
	.8	1.524	1.128	1.090	1.316
	.9	1.691	1.162	1.117	1.404
	1.0	1.893	1.200	1.141	1.516
5,000	0	0.8318	0.9657	0.9753	0.8679
	.1	.8374	.9676	.9764	.8718
	.2	.8554	.9734	.9809	.8839
	.3	.8852	.9830	.9875	.9041
	.4	.9291	.9965	.9973	.9333
	.5	.9868	1.014	1.010	.9703
	.6	1.061	1.035	1.025	1.018
	.7	1.154	1.060	1.044	1.073
	.8	1.268	1.089	1.064	1.141
	.9	1.407	1.122	1.086	1.223
	1.0	1.575	1.159	1.117	1.309
10,000	0	0.6881	0.9312	0.9491	0.7513
	.1	.6925	.9331	.9504	.7541
	.2	.7075	.9387	.9549	.7647
	.3	.7320	.9480	.9621	.7814
	.4	.7684	.9609	.9714	.8069
	.5	.8157	.9776	.9836	.8388
	.6	.8776	.9983	.9989	.8794
	.7	.9542	1.022	1.016	.9291
	.8	1.048	1.050	1.037	.9859
	.9	1.163	1.082	1.058	1.057
	1.0	1.302	1.117	1.083	1.137
15,000	0	0.5643	0.8969	0.9223	0.6461
	.1	.5681	.8987	.9233	.6490
	.2	.5799	.9040	.9281	.6572
	.3	.6002	.9131	.9347	.6720
	.4	.6300	.9256	.9448	.6931
	.5	.6692	.9416	.9570	.7206
	.6	.7198	.9615	.9719	.7553
	.7	.7826	.9848	.9891	.7973
	.8	.8601	1.012	1.008	.8482
	.9	.9542	1.042	1.031	.9062
	1.0	1.068	1.076	1.055	.9762
20,000	0	0.4596	0.8626	0.8960	0.5523
	.1	.4629	.8644	.8966	.5553
	.2	.4726	.8696	.9016	.5622
	.3	.4891	.8780	.9072	.5754
	.4	.5132	.8902	.9172	.5930
	.5	.5454	.9058	.9289	.6170
	.6	.5865	.9247	.9440	.6461
	.7	.6375	.9470	.9610	.6817
	.8	.7004	.9728	.9798	.7248
	.9	.7769	1.002	1.002	.7746
	1.0	.8700	1.035	1.026	.8341
25,000	0	0.3710	0.8281	0.8682	0.4696
	.1	.3737	.8299	.8700	.4715
	.2	.3814	.8347	.8740	.4776
	.3	.3948	.8430	.8804	.4884
	.4	.4145	.8545	.8891	.5043
	.5	.4399	.8696	.9016	.5233
	.6	.4731	.8877	.9151	.5487
	.7	.5147	.9092	.9316	.5794
	.8	.5667	.9339	.9515	.6152
	.9	.6276	.9620	.9724	.6581
	1.0	.7023	.9934	.9950	.7092
30,000	0	0.2968	0.7938	0.8414	0.3959
	.1	.2989	.7954	.8430	.3975
	.2	.3052	.8002	.8469	.4029
	.3	.3158	.8081	.8525	.4121
	.4	.3315	.8195	.8621	.4248
	.5	.3519	.8335	.8727	.4416
30,000	0.6	0.3787	0.8509	0.8862	0.4633
	.7	.4118	.8715	.9029	.4886
	.8	.4572	.8954	.9207	.5190
	.9	.5019	.9222	.9416	.5551
	1.0	.5619	.9524	.9655	.5964
35,000	0	0.2352	0.7595	0.8149	0.3312
	.1	.2368	.7611	.8164	.3325
	.2	.2418	.7655	.8196	.3372
	.3	.2502	.7732	.8257	.3446
	.4	.2627	.7838	.8337	.3559
	.5	.2789	.7975	.8443	.3699
	.6	.3001	.8141	.8576	.3878
	.7	.3262	.8339	.8727	.4093
	.8	.3583	.8568	.8910	.4345
	.9	.3977	.8825	.9111	.4647
	1.0	.4452	.9112	.9334	.4997
40,000	0	0.1853	0.7572	0.8130	0.2619
	.1	.1866	.7588	.8141	.2631
	.2	.1905	.7632	.8176	.2667
	.3	.1972	.7709	.8239	.2726
	.4	.2070	.7815	.8321	.2814
	.5	.2198	.7950	.8430	.2924
	.6	.2364	.8118	.8562	.3065
	.7	.2570	.8314	.8714	.3236
	.8	.2824	.8539	.8889	.3438
	.9	.3134	.8798	.9090	.3676
	1.0	.3506	.9085	.9310	.3951
45,000	0	0.1459	0.7572	0.8130	0.2062
	.1	.1469	.7588	.8141	.2071
	.2	.1500	.7632	.8176	.2100
	.3	.1552	.7709	.8239	.2145
	.4	.1630	.7815	.8321	.2216
	.5	.1730	.7950	.8430	.2302
	.6	.1862	.8118	.8562	.2414
	.7	.2024	.8314	.8714	.2548
	.8	.2224	.8539	.8889	.2708
	.9	.2467	.8798	.9090	.2894
	1.0	.2762	.9085	.9310	.3112
50,000	0	0.1149	0.7572	0.8130	0.1624
	.1	.1157	.7588	.8141	.1631
	.2	.1181	.7632	.8176	.1654
	.3	.1223	.7709	.8239	.1691
	.4	.1284	.7815	.8321	.1746
	.5	.1362	.7950	.8430	.1812
	.6	.1466	.8118	.8562	.1900
	.7	.1594	.8314	.8714	.2006
	.8	.1751	.8539	.8889	.2132
	.9	.1943	.8798	.9090	.2279
	1.0	.2175	.9085	.9310	.2451
55,000	0	0.0905	0.7572	0.8130	0.1279
	.1	.0911	.7588	.8141	.1285
	.2	.0930	.7632	.8175	.1302
	.3	.0963	.7709	.8239	.1331
	.4	.1011	.7815	.8321	.1374
	.5	.1073	.7950	.8430	.1428
	.6	.1155	.8118	.8562	.1497
	.7	.1255	.8314	.8714	.1580
	.8	.1379	.8539	.8889	.1679
	.9	.1530	.8798	.9090	.1795
	1.0	.1713	.9085	.9310	.1930
60,000	0	0.0713	0.7572	0.8130	0.1008
	.1	.0717	.7588	.8141	.1011
	.2	.0733	.7632	.8175	.1026
	.3	.0758	.7709	.8239	.1048
	.4	.0796	.7815	.8321	.1082
	.5	.0845	.7950	.8430	.1124
	.6	.0909	.8118	.8562	.1178
	.7	.0988	.8314	.8714	.1244
	.8	.1086	.8539	.8889	.1322
	.9	.1205	.8798	.9090	.1413
	1.0	.1349	.9085	.9310	.1520

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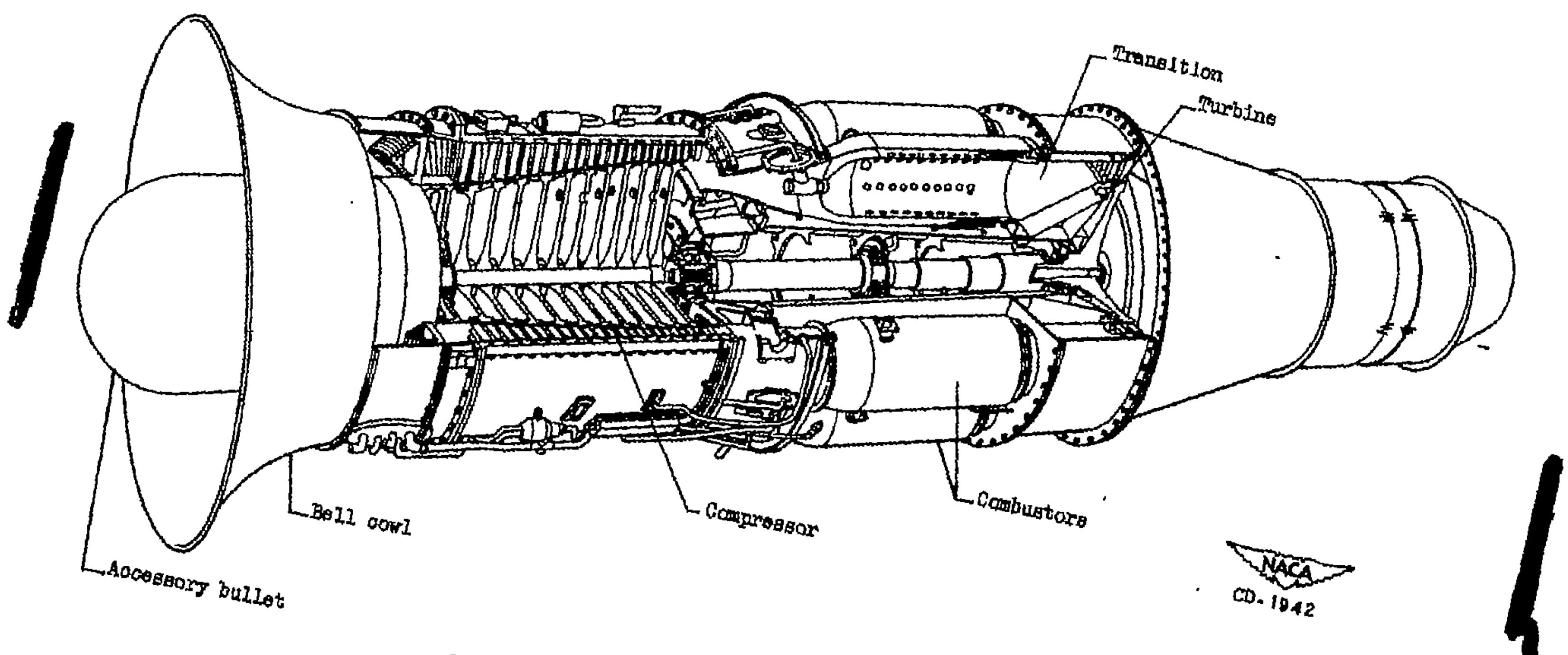

Figure 1. - Cutaway view of J35-A-29 engine.

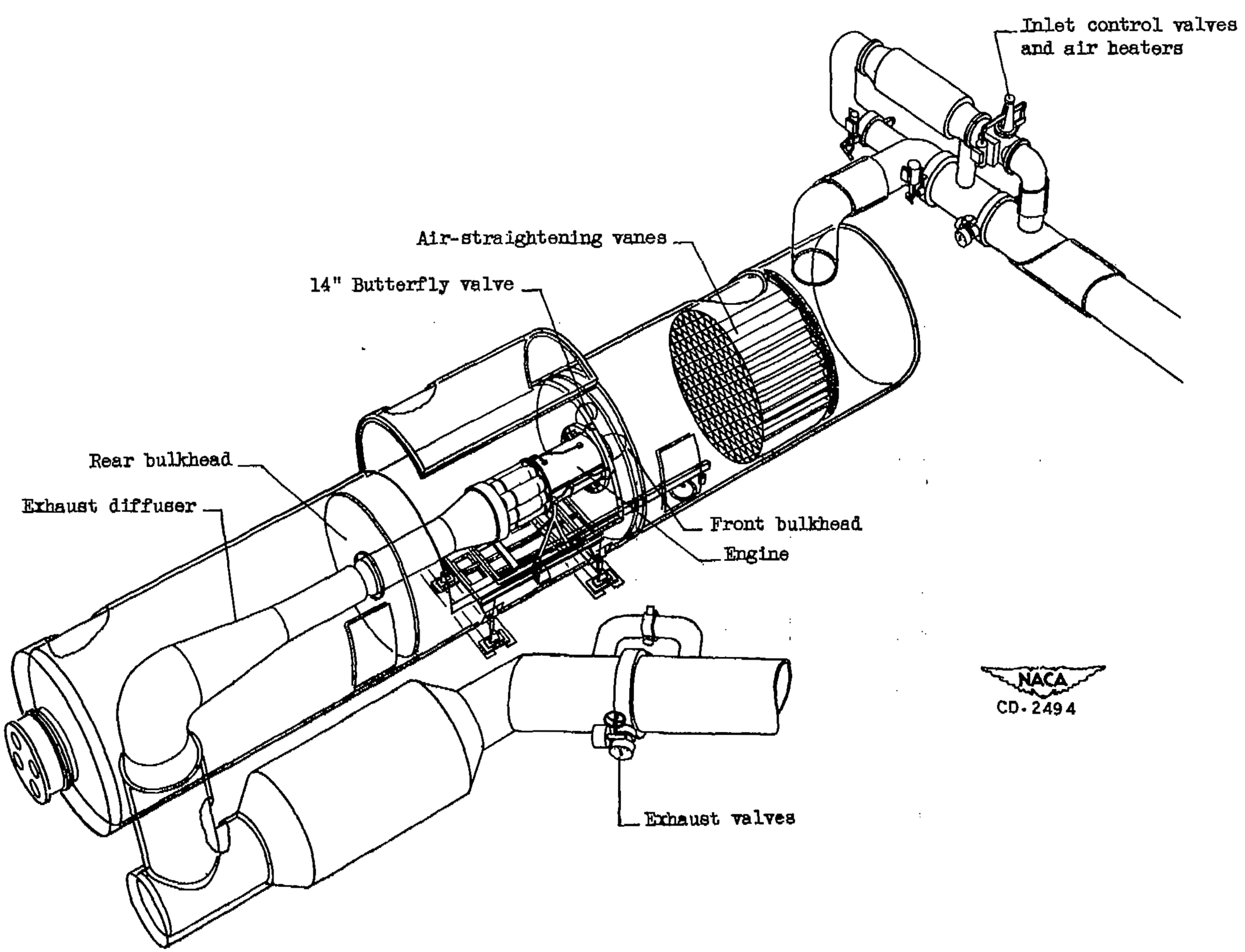

Figure 2. - Altitude chamber with J35-A-29 engine installed in test section.

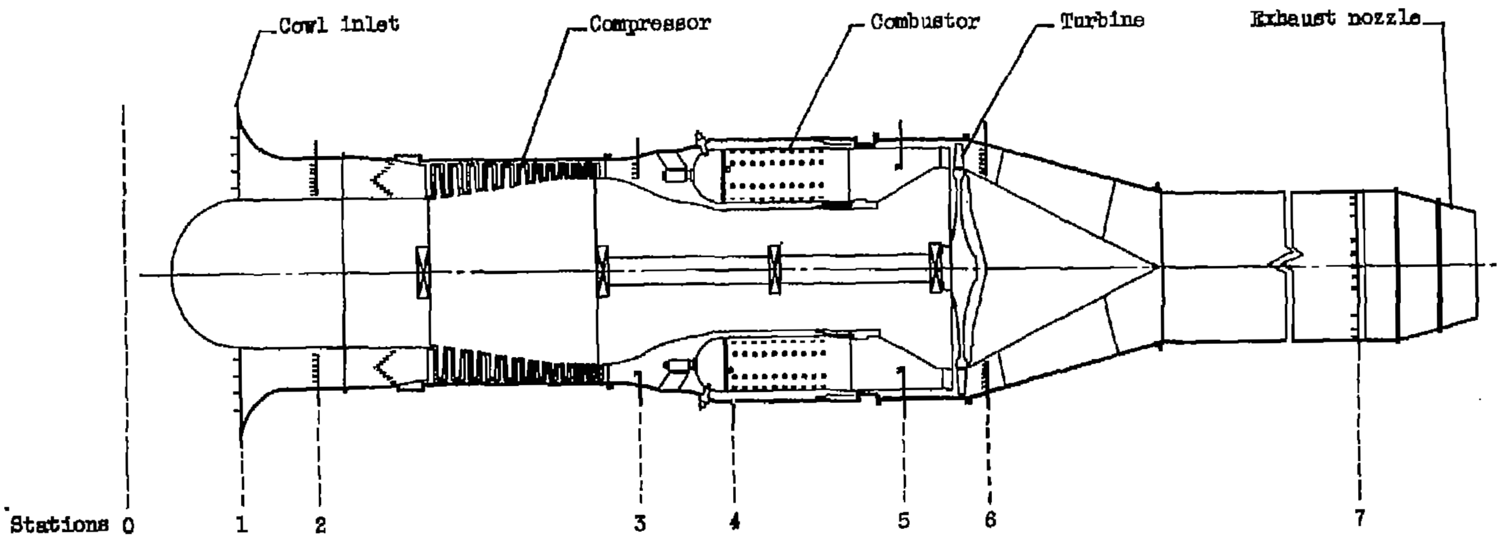

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Figure 3. - Location of instrumentation stations.

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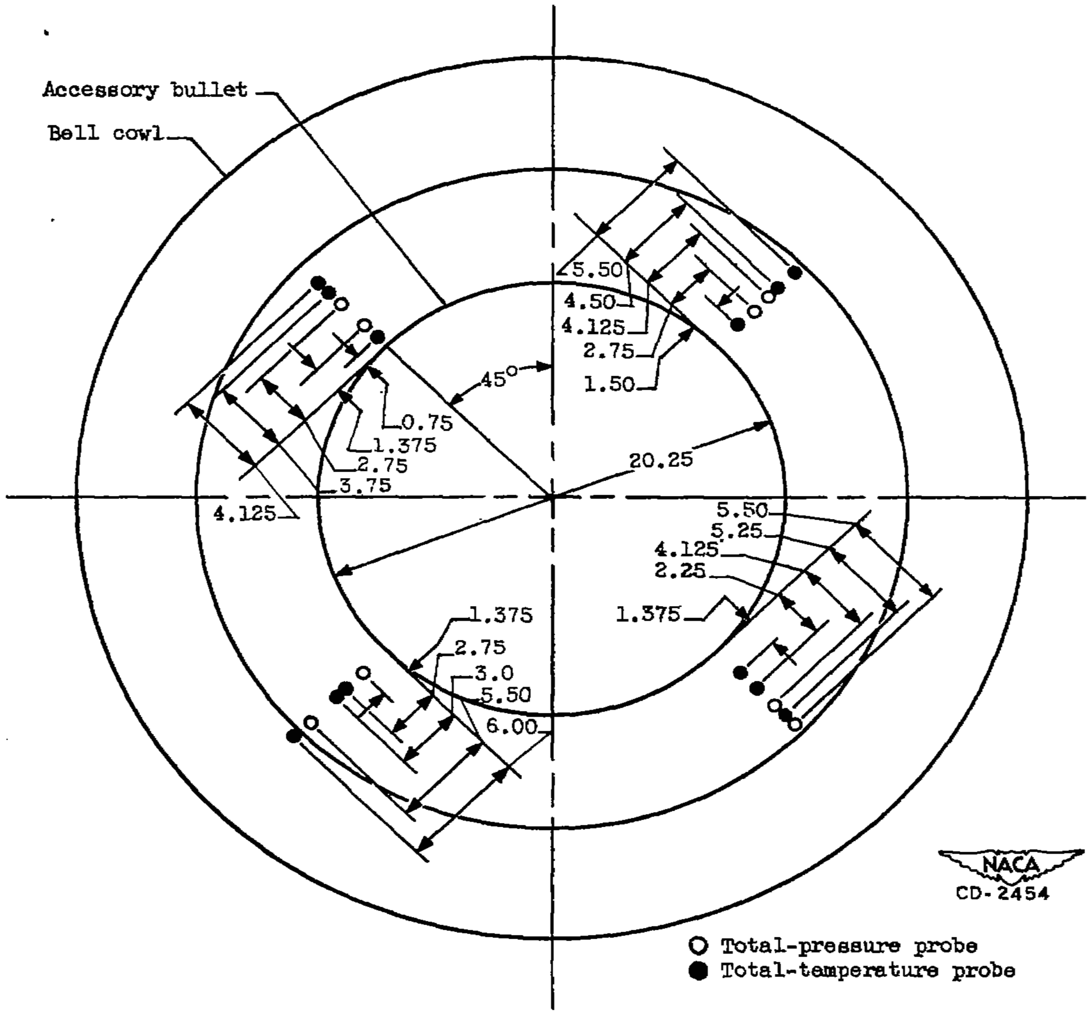
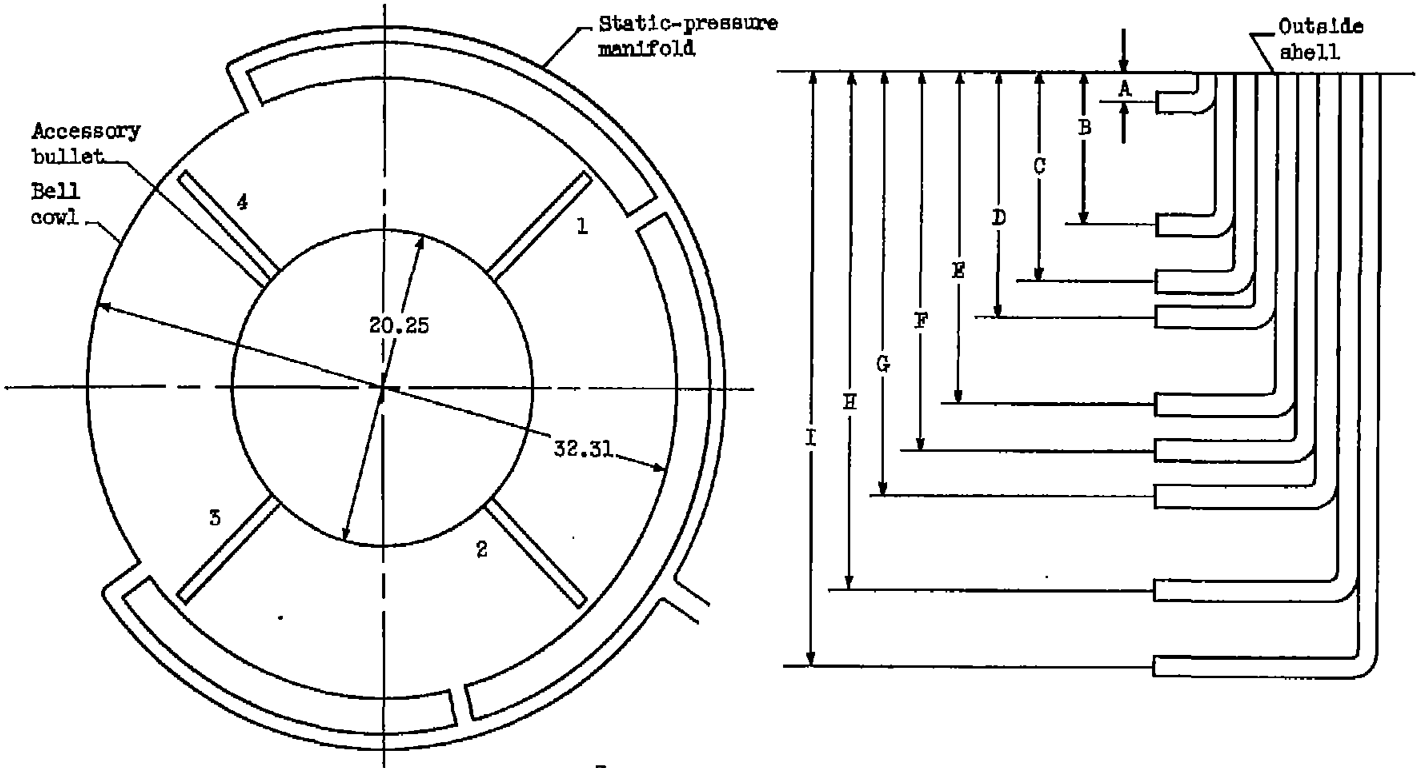


(a) Station 1, engine inlet.

Figure 4. - Details of instrumentation. (All dimensions in inches.)

[redacted]



[P and T indicate total pressure and total temperature probes, respectively.]

	A	B	C	D	E	F	G	H	I
Rake 1	0.17	1.01	1.56	1.91	2.86	3.49	3.89	5.01	5.77
Type of probe	P	P	T	P	P	T	P	P	T
Rake 2	0.37	0.69	1.23	2.14	2.49	3.11	4.16	4.58	5.31
Type of probe	P	T	P	P	T	P	P	T	P
Rake 3	0.58	1.12	1.45	2.37	2.98	3.36	4.44	5.16	5.62
Type of probe	P	T	P	P	T	P	P	T	P
Rake 4	0.27	0.79	1.68	2.02	2.61	3.62	4.02	4.72	5.93
Type of probe	T	P	P	T	P	P	T	P	P

(b) Station 2, compressor inlet.

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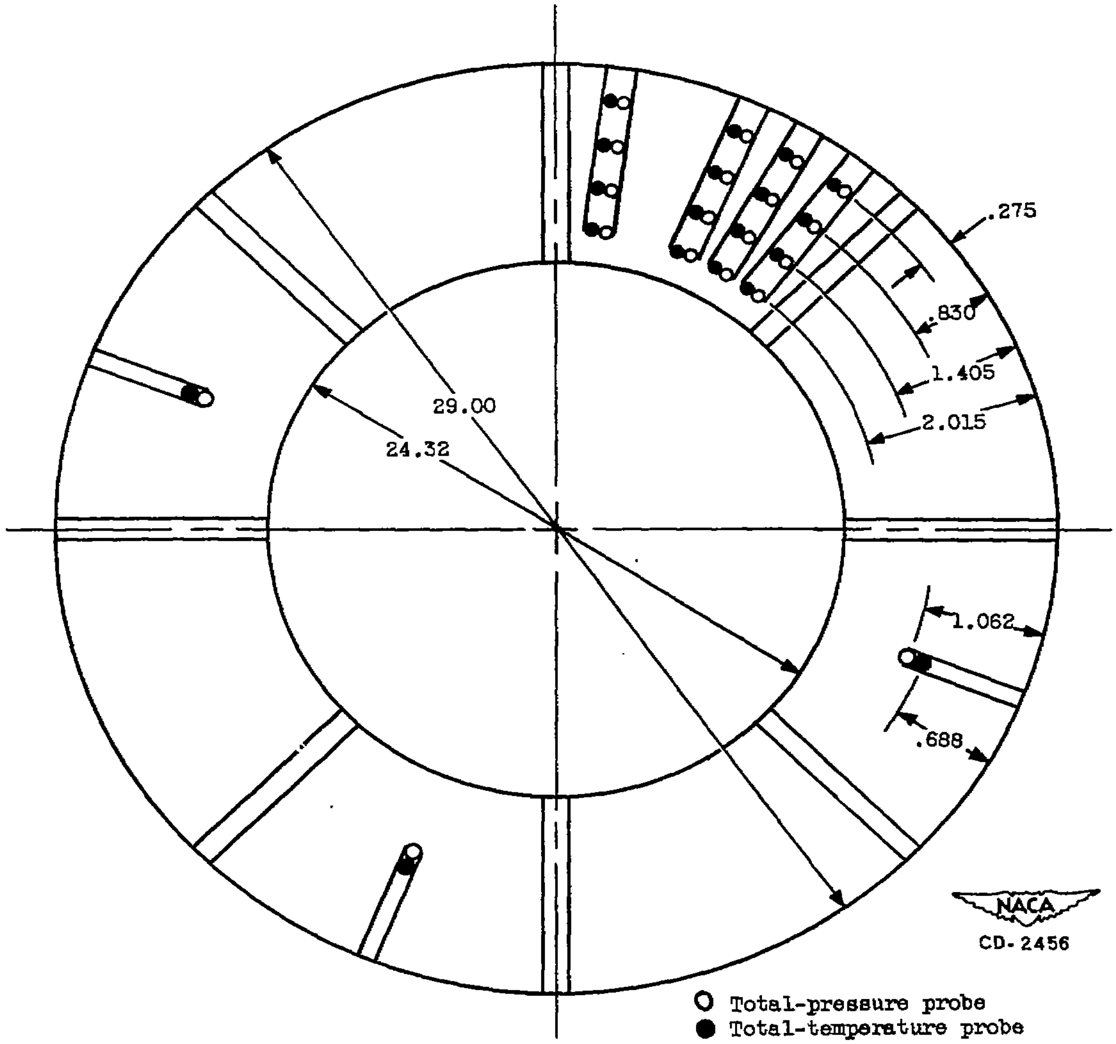
Figure 4. - Continued. Details of instrumentation. (All dimensions in inches.)



(c) Station 3, compressor discharge.

Figure 4. - Continued. Details of instrumentation. (All dimensions in inches.)

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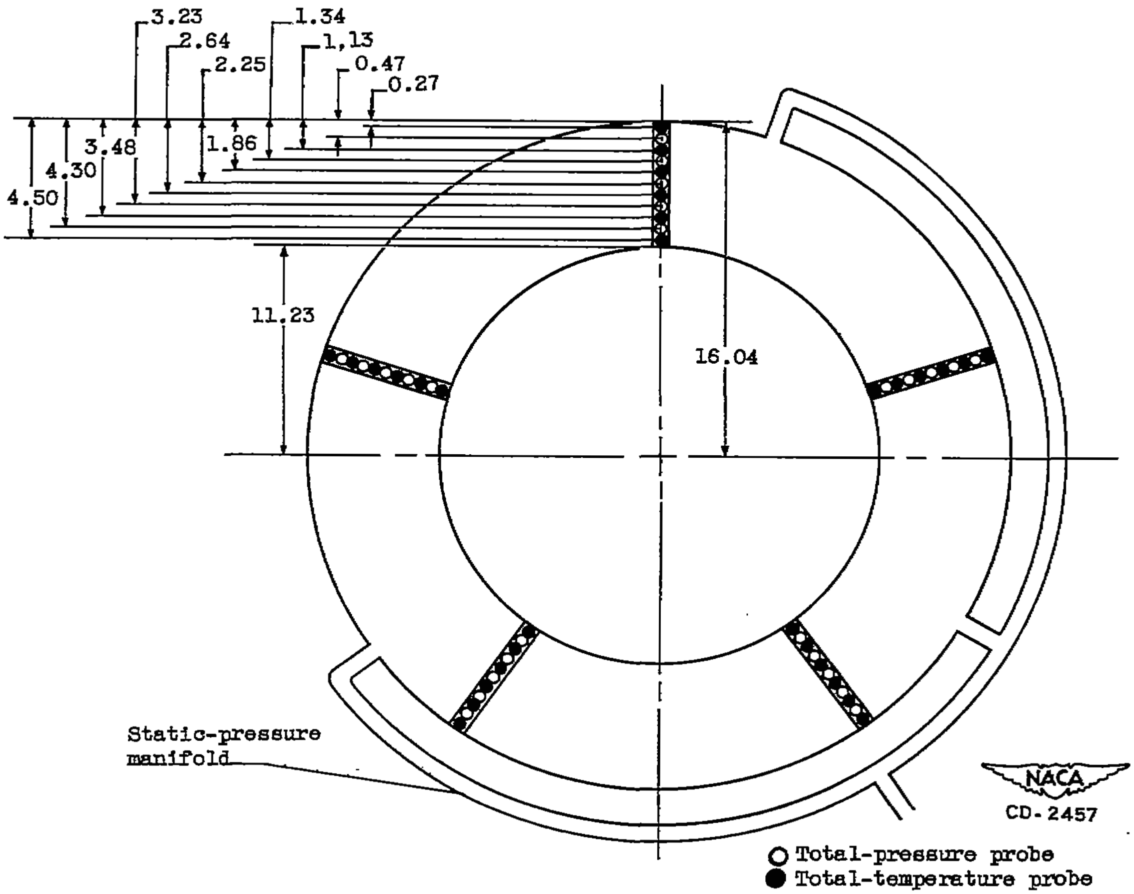
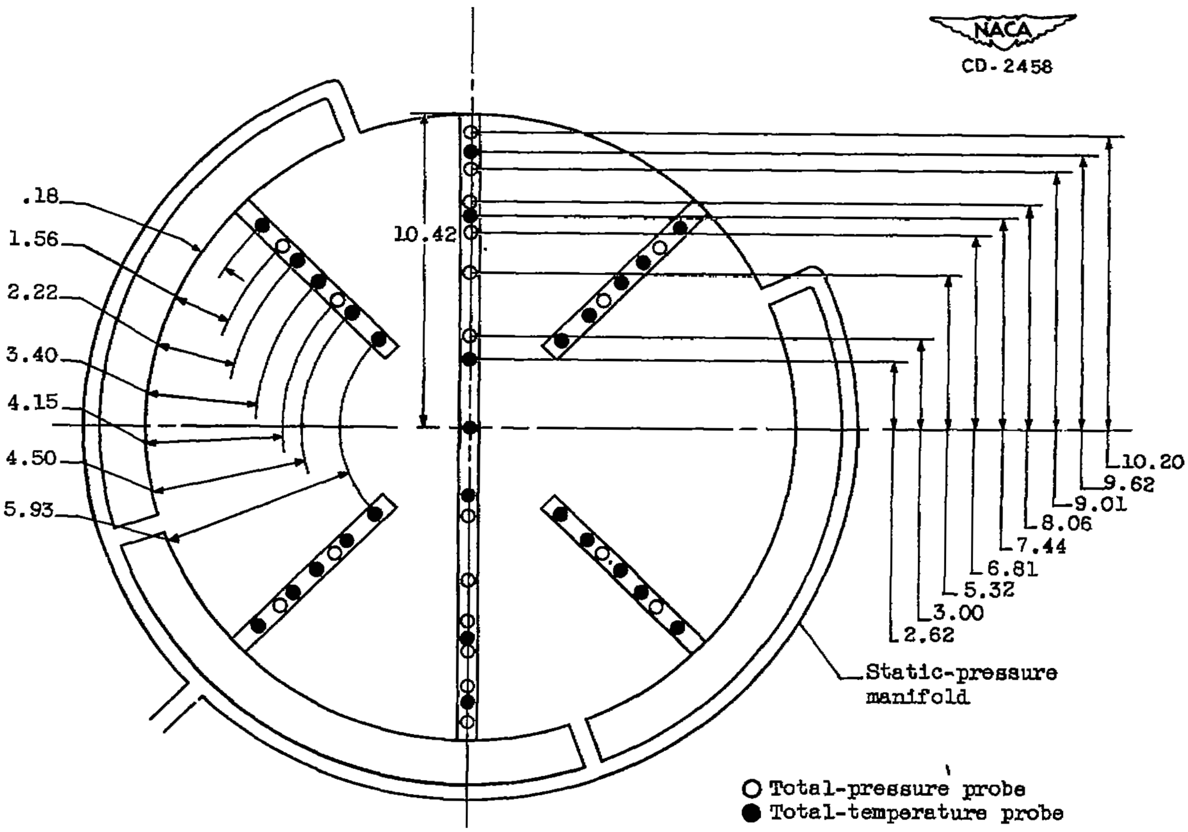


(d) Station 6, turbine discharge.

Figure 4. - Continued. Details of instrumentation. (All dimensions in inches.)



(e) Station 7, exhaust-nozzle inlet.

Figure 4. - Concluded. Details of instrumentation. (All dimensions in inches.)

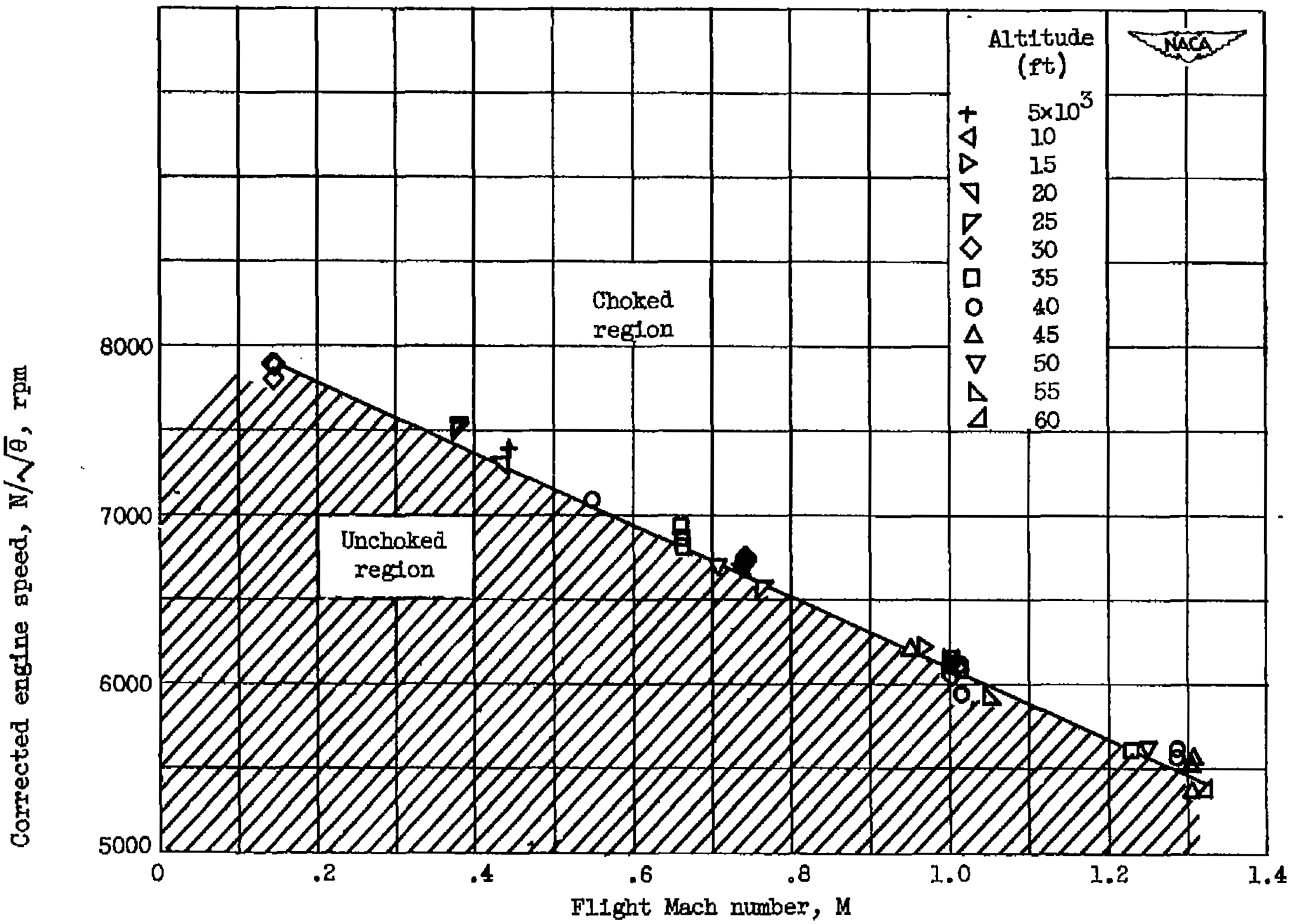


Figure 5. - Minimum corrected engine speed at which critical flow existed in exhaust nozzle.

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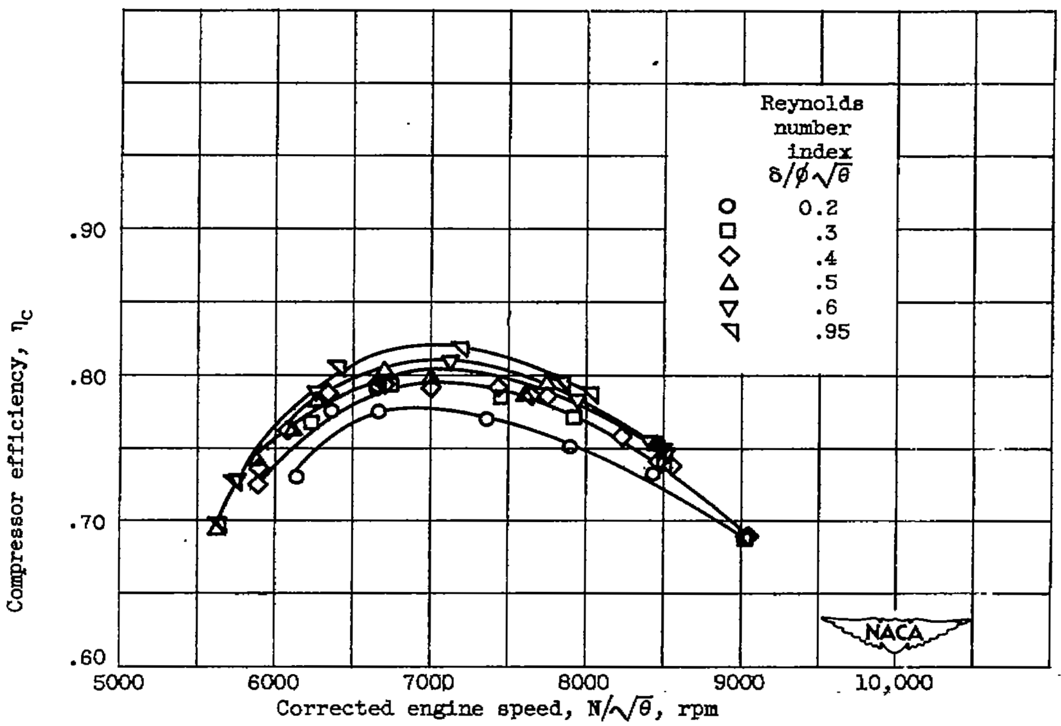

(a) Variation of compressor efficiency with corrected engine speed.

Figure 6. - Effect of Reynolds number index on compressor performance characteristics. (Reynolds number index at sea-level conditions, 1.0.)

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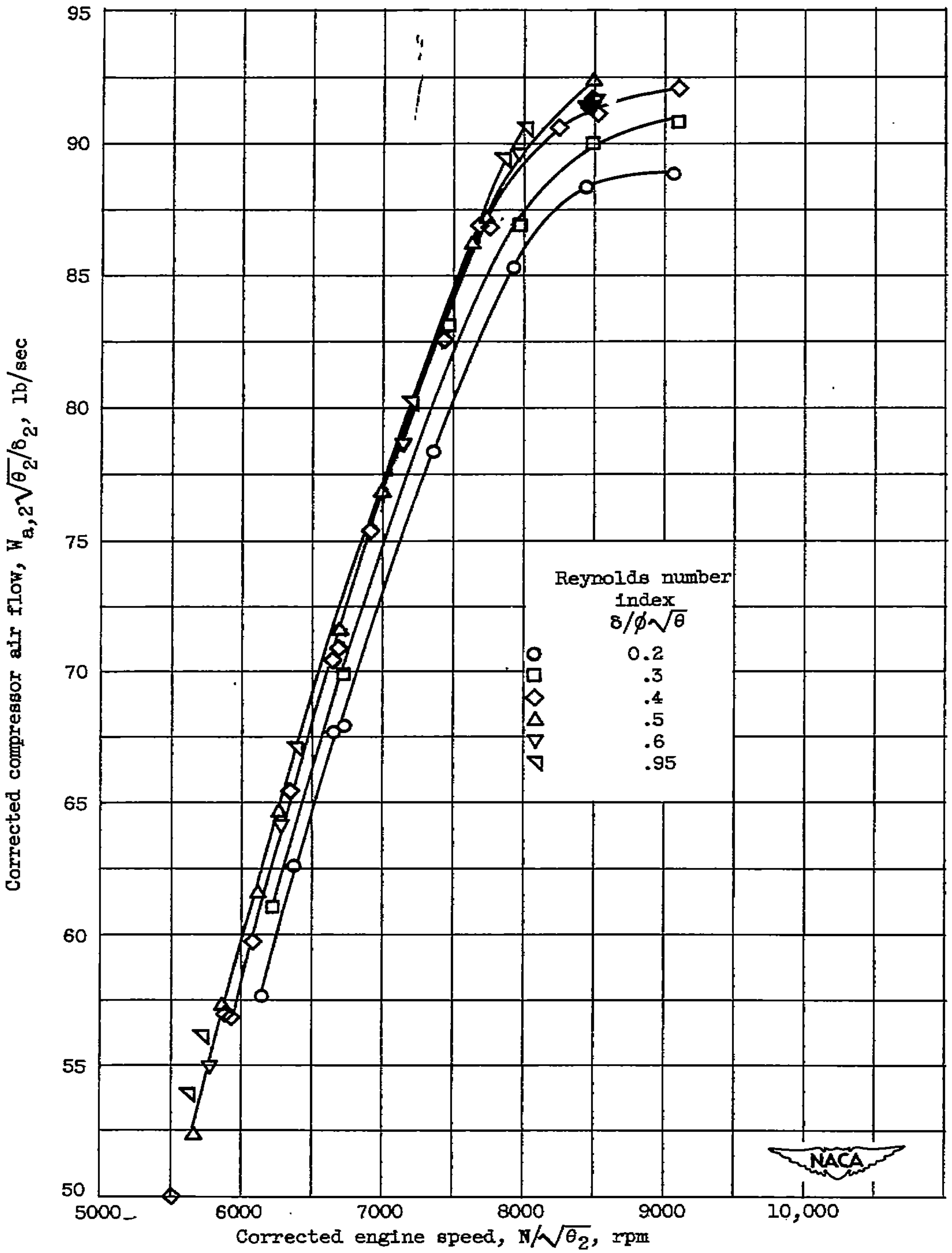

(b) Variation of corrected compressor air flow with engine speed.

Figure 6. - Continued. Effect of Reynolds number index on compressor performance characteristics. (Reynolds number index at sea-level static conditions, 1.0.)

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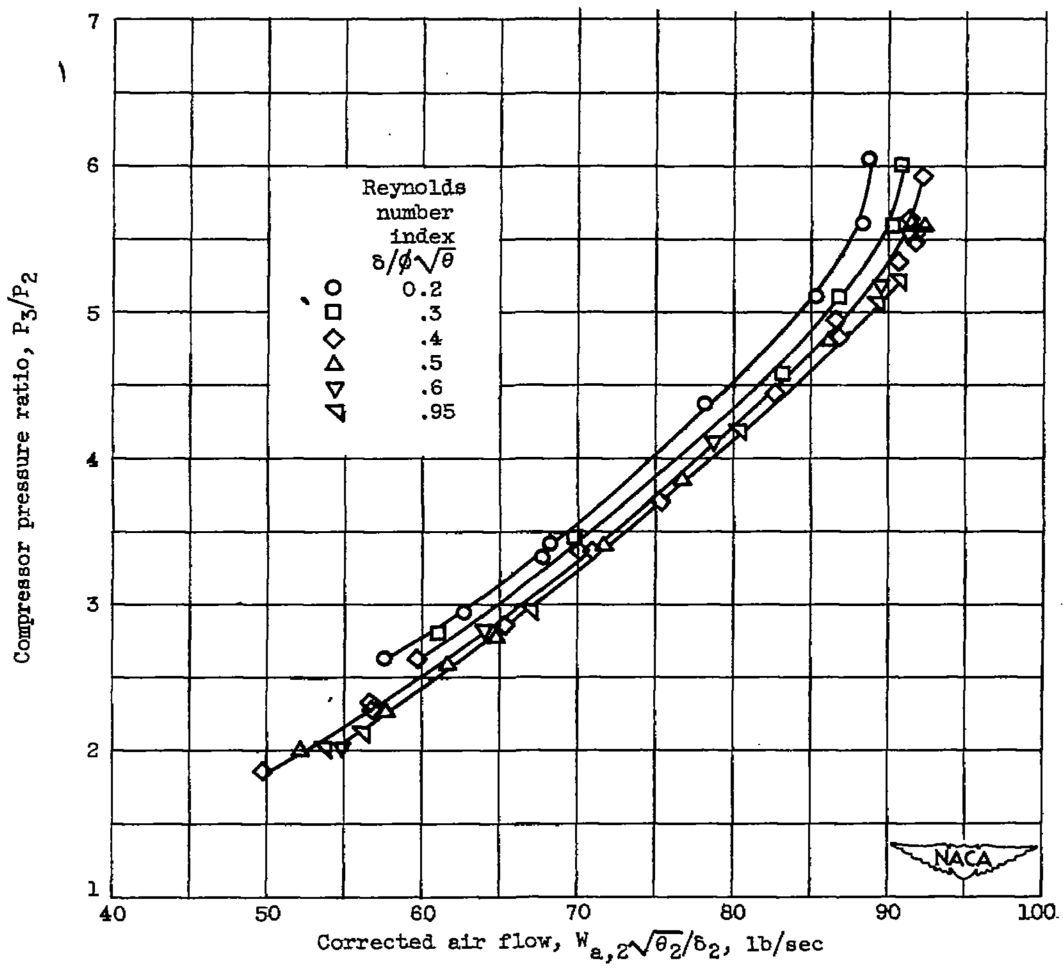

(c) Variation of compressor pressure ratio with corrected air flow.

Figure 6. - Continued. Effect of Reynolds number index on compressor performance characteristics. (Reynolds number index at sea-level static conditions, 1.0.)

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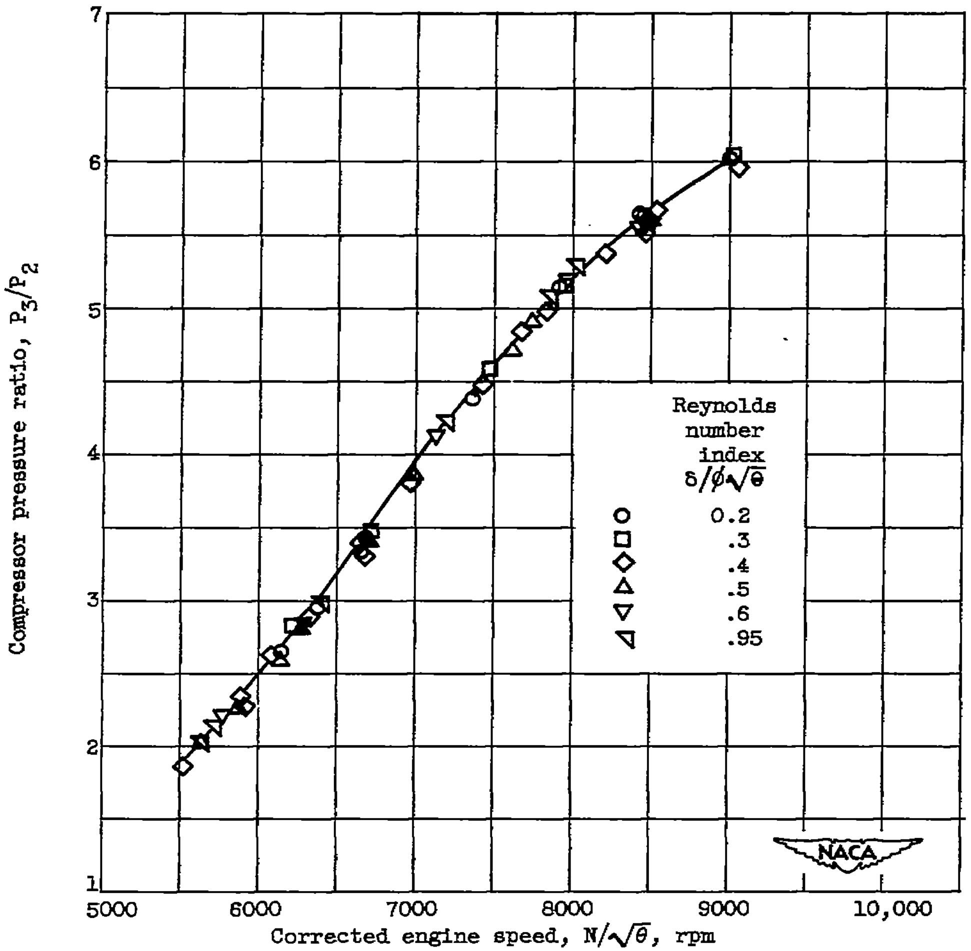


(d) Variation of compressor pressure ratio with corrected engine speed.

Figure 6. - Concluded. Effect of Reynolds number index on compressor performance characteristics. (Reynolds number index at sea-level conditions, 1.0.)

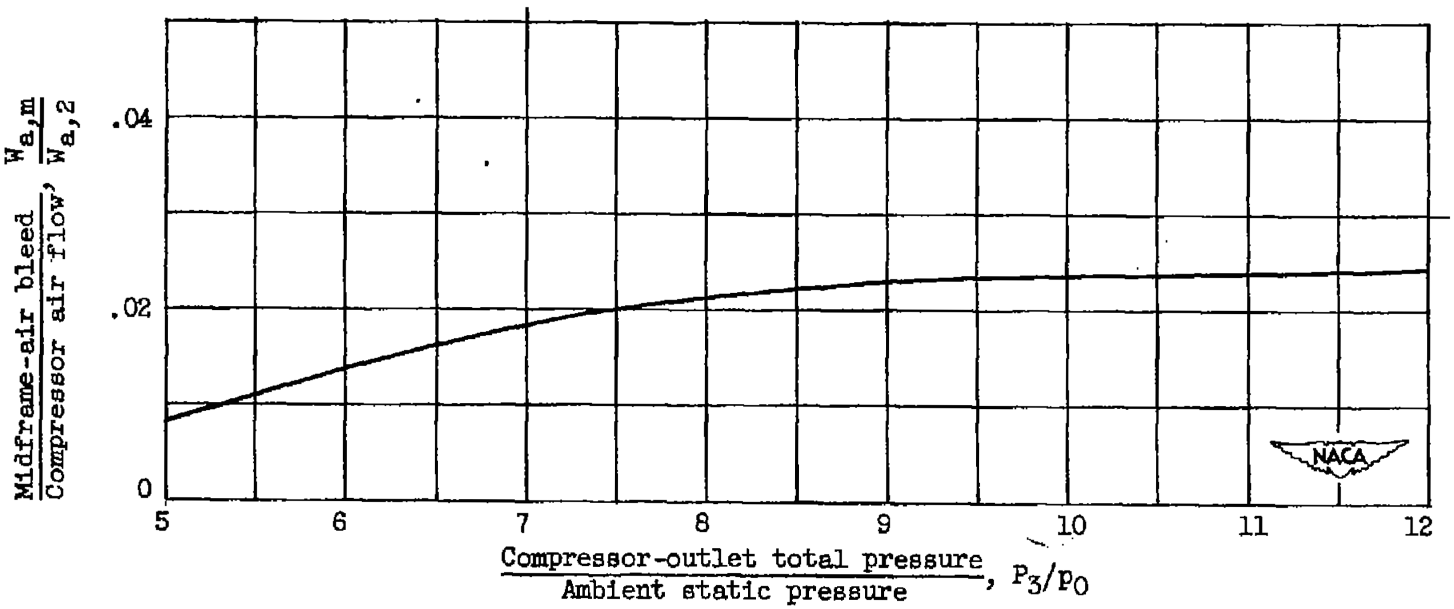
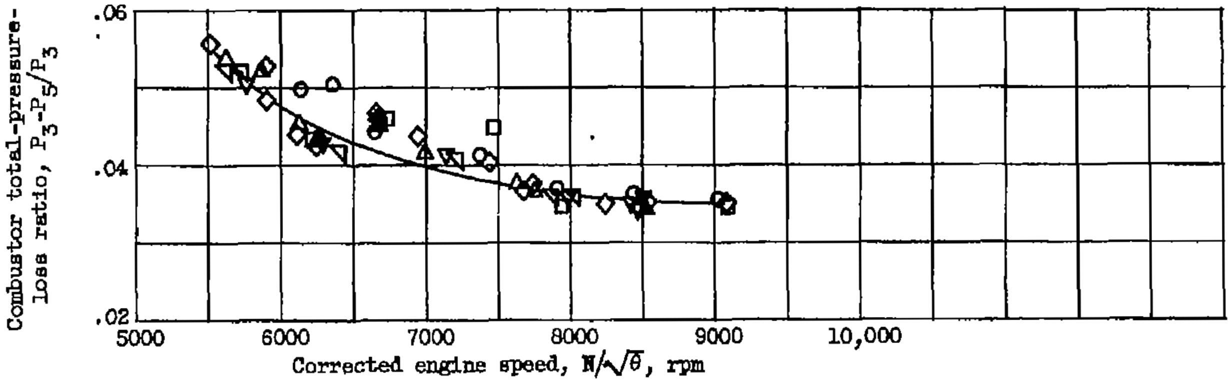
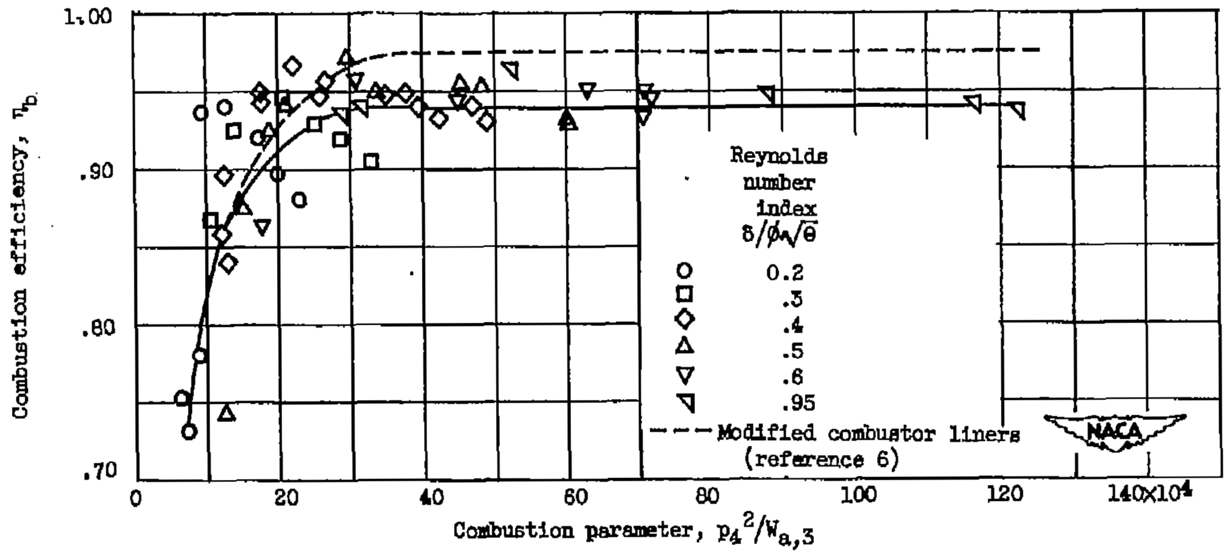


Figure 7. - Midframe air bleed.

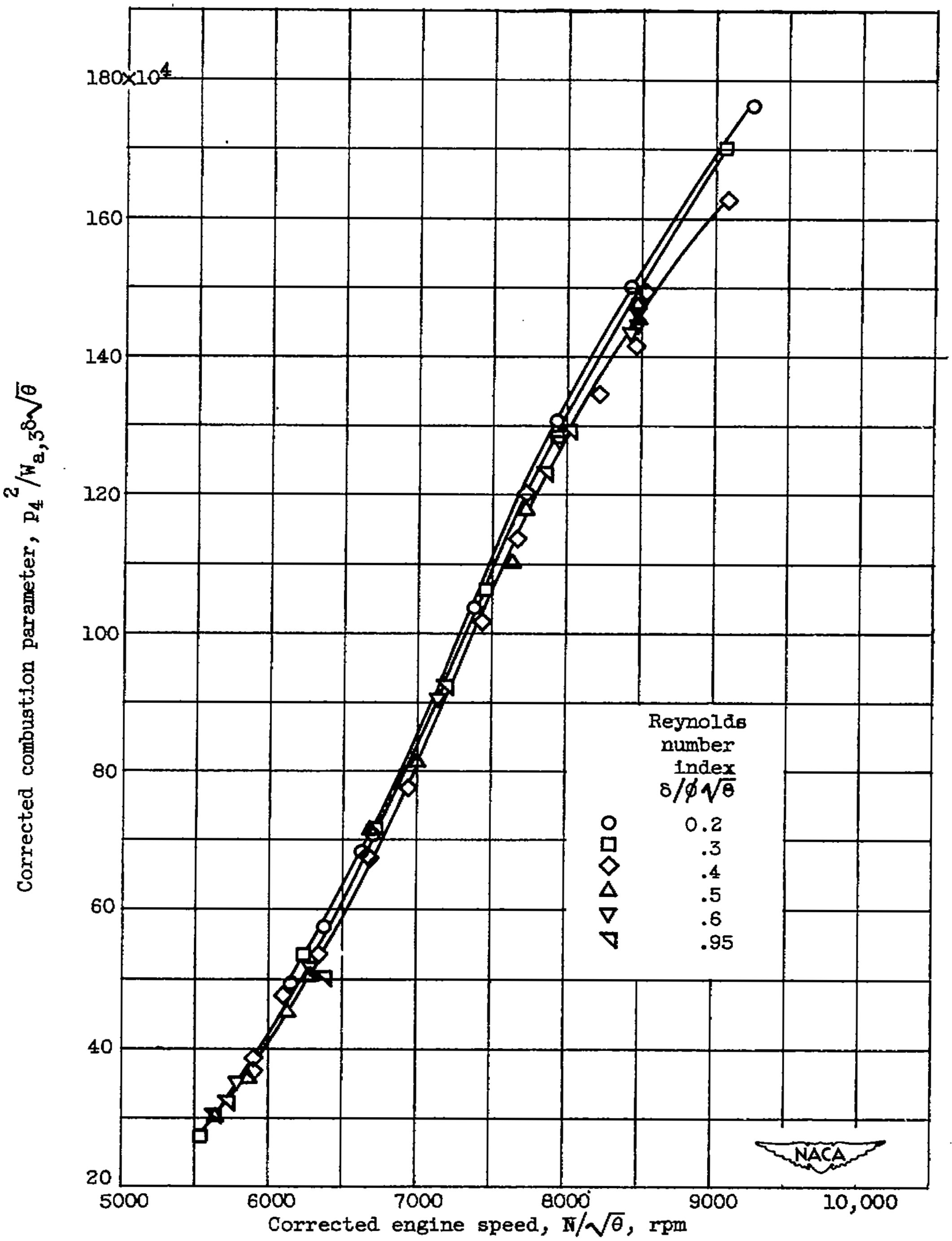


(a) Variation of combustor total-pressure-loss ratio with corrected engine speed.



(b) Variation of combustion efficiency with combustion parameter.

Figure 8. - Continued. Effect of Reynolds number index on combustor performance. (Reynolds number index at sea-level static conditions, 1.0.)



(c) Variation of corrected combustion parameter with corrected engine speed.

Figure 8. - Concluded. Effect of Reynolds number index on combustor performance. (Reynolds number index at sea-level static conditions, 1.0.)

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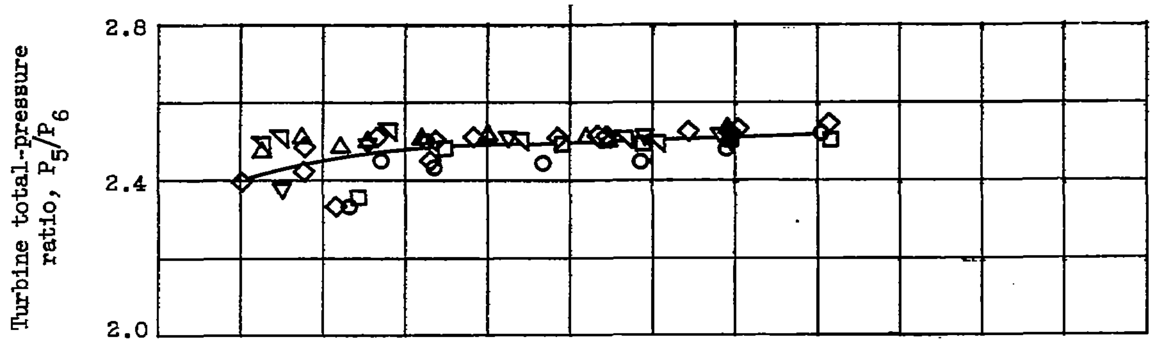
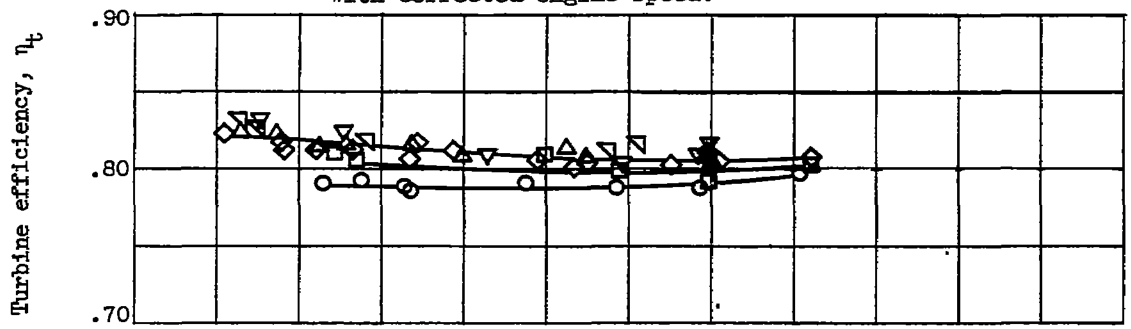
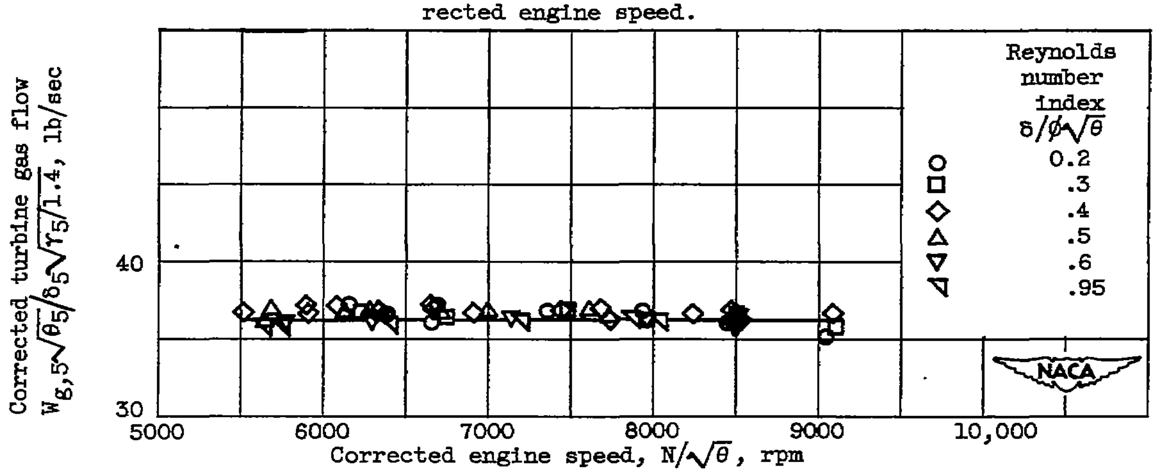


(a) Variation of turbine total-pressure ratio with corrected engine speed.



(b) Variation of turbine efficiency with corrected engine speed.



(c) Variation of corrected turbine gas flow with corrected engine speed.

Figure 9. - Effect of Reynolds number index on turbine performance. (Reynolds number index at sea-level static conditions, 1.0.)

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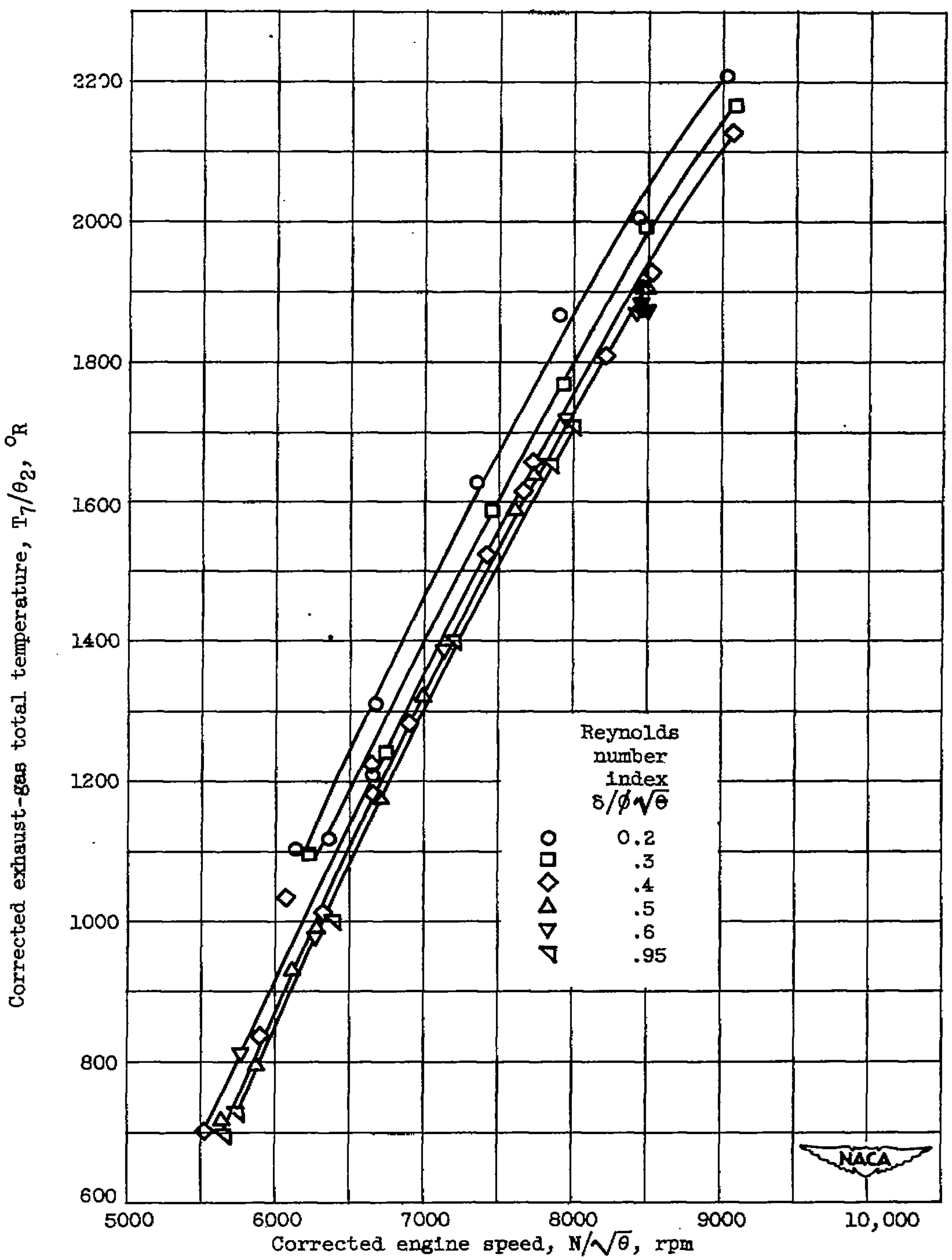

(a) Variation of corrected exhaust-gas total temperature with corrected engine speed.

Figure 10. - Effect of Reynolds number index on generalized engine performance. (Reynolds number index at sea-level static conditions, 1.0.)

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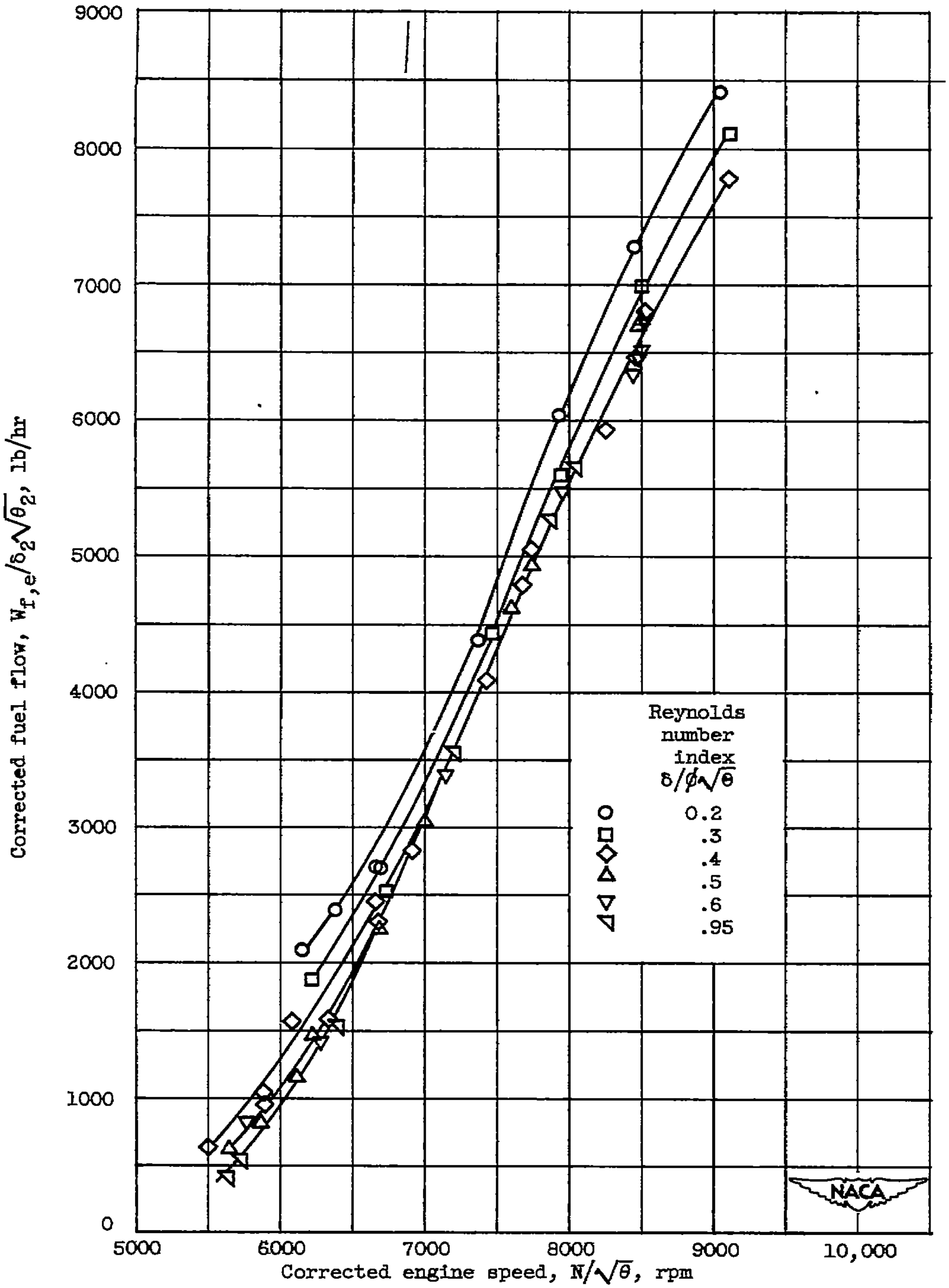

(b) Variation of corrected fuel flow with corrected engine speed.

Figure 10. - Continued. Effect of Reynolds number index on generalized engine performance. (Reynolds number index at sea-level static conditions, 1.0.)

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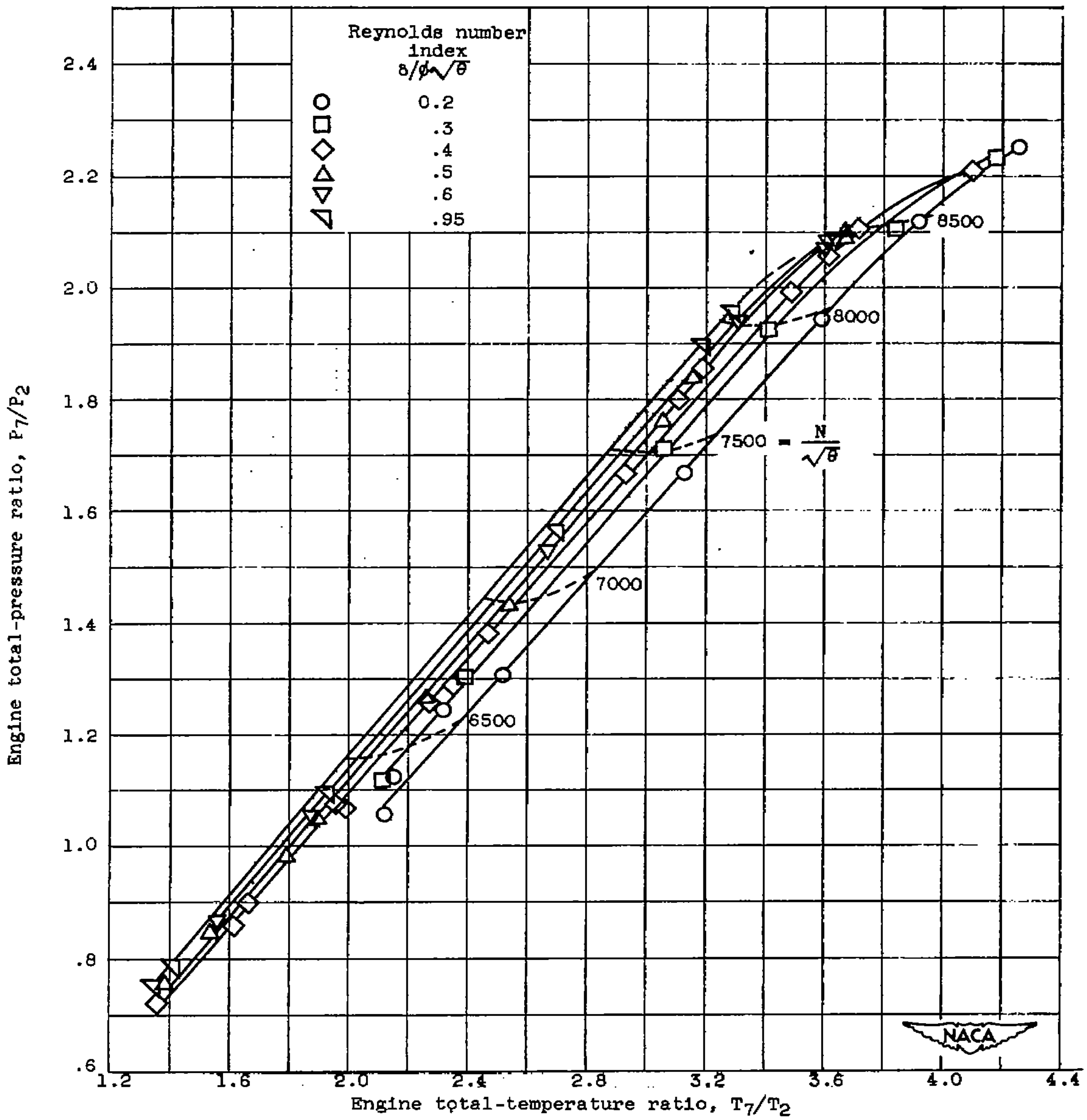
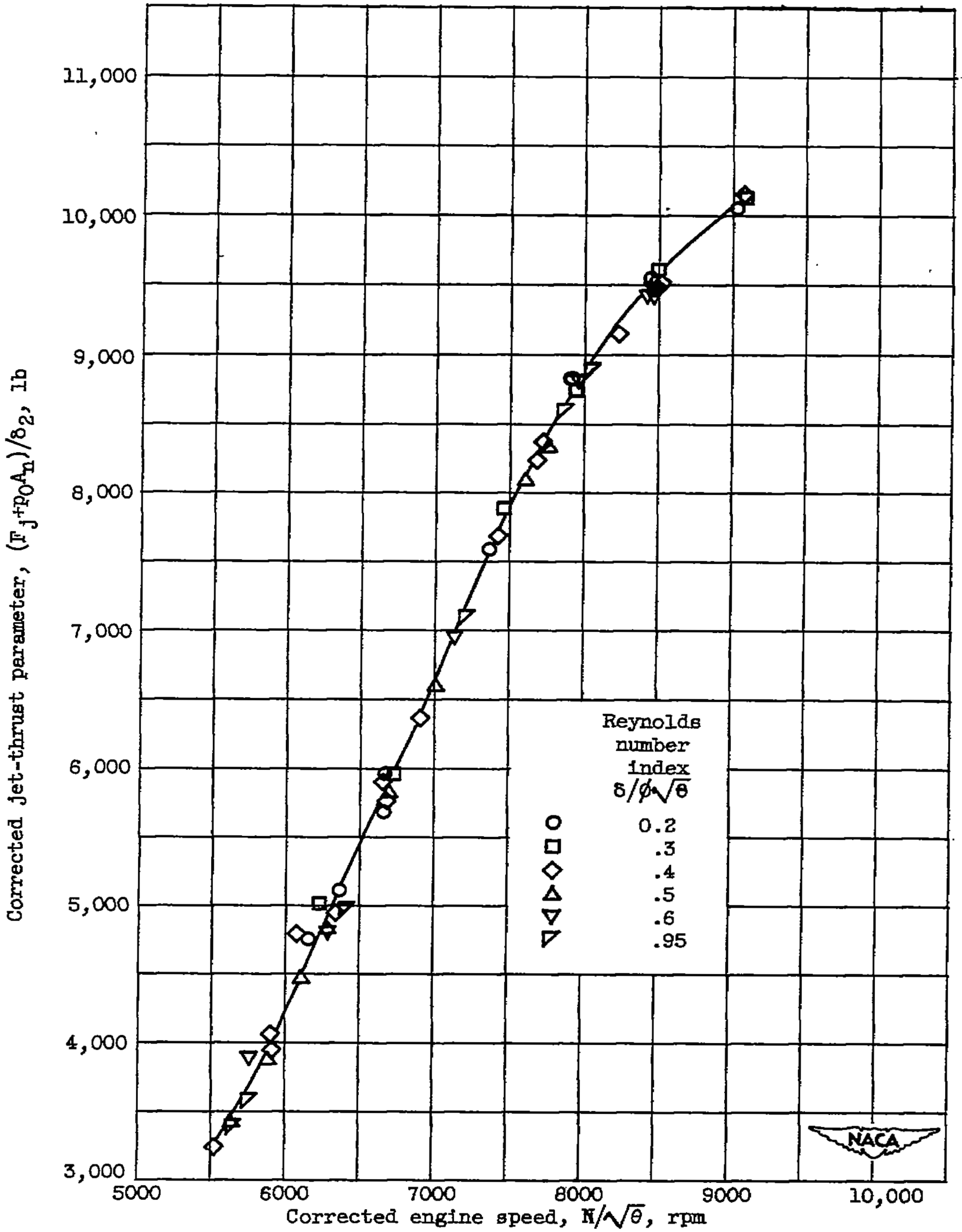


(c) Engine pumping characteristics.

Figure 10. - Continued. Effect of Reynolds number index on generalized engine performance.



(d) Variation of corrected jet-thrust parameter with corrected engine speed.

Figure 10. - Concluded. Effect of Reynolds number index on generalized engine performance. (Reynolds number index at sea-level static conditions, 1.0.)

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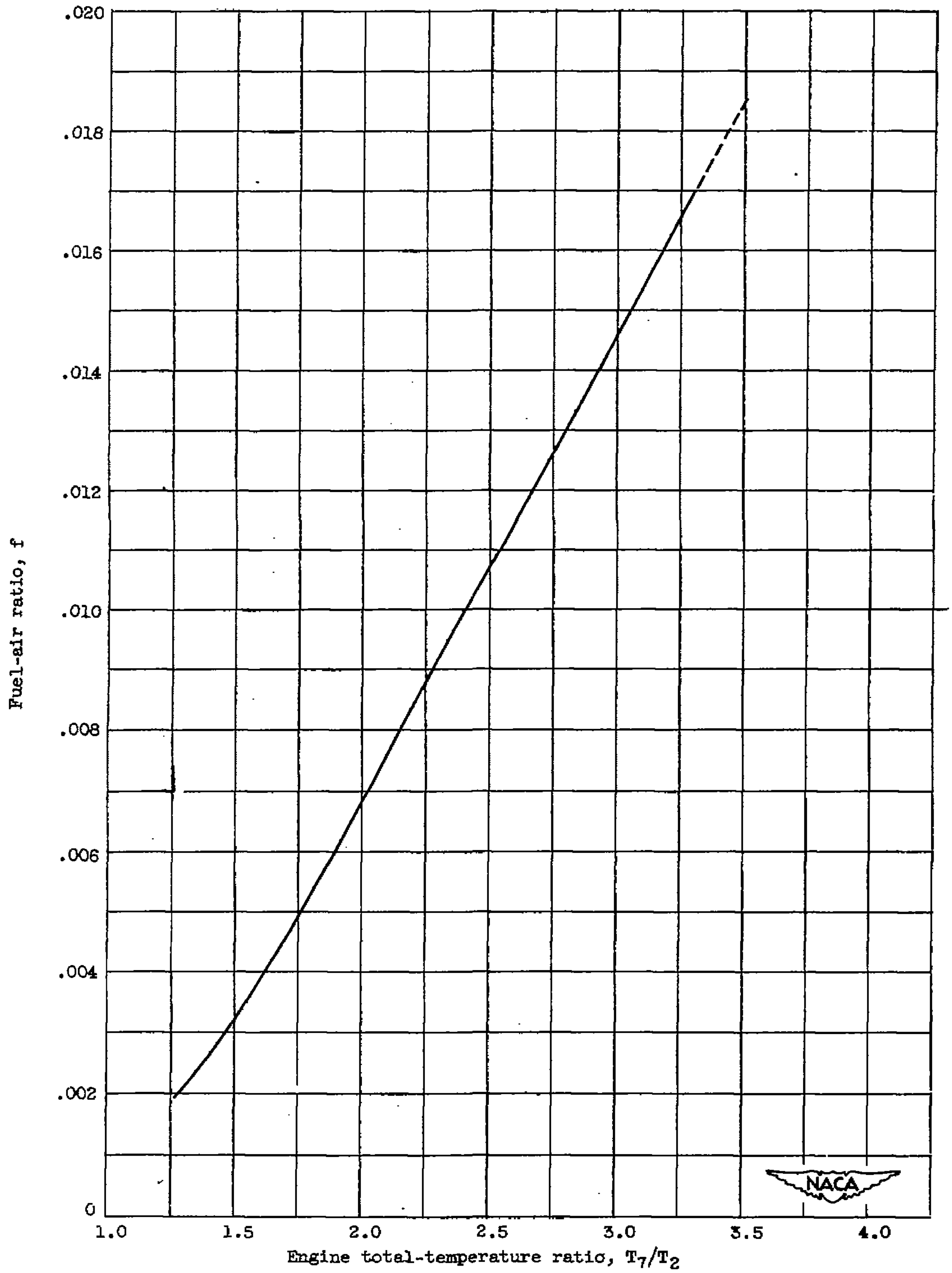

Figure 11. - Variation of fuel-air ratio with engine total-temperature ratio for standard sea-level inlet conditions.

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